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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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AGARD Conference Proceedings No.388

## Guidance, Control and Positioning of Future Precision Guided Stand-Off Weapons Systems

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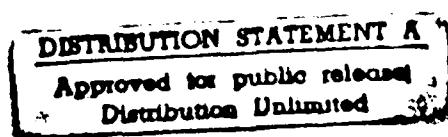


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AGARD Conference Proceedings No.388  
**GUIDANCE, CONTROL AND POSITIONING OF FUTURE PRECISION  
GUIDED STAND-OFF WEAPONS SYSTEMS**

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Published June 1986

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ISBN 92-835-0390-2



Printed by Specialised Printing Services Limited  
40 Chigwell Lane, Loughton, Essex IG10 3TZ

## PREFACE

The environment in which tactical air forces must be able to operate is becoming increasingly more lethal. Aircraft will face highly defended targets which they must approach through a hostile network of integrated air defences. These difficult defences provide strong motivation to perform some particular missions by using stand-off unmanned weapon systems. Such stand-off weapons may be air launched or ground launched. In either case the attrition associated with penetration and terminal threats will be reduced. Also, the use of extended range stand-off missiles can increase the effective operational range of the carrying aircraft.

Application of stand-off weapons places increased emphasis on mission planning and communication functions. Planning data must be made available on appropriate time lines. Decision criteria must be established and integrated for utilization of more expensive stand-off weapon options. Unmanned stand-off weapons will require improved sensors for guidance and control, positioning, and navigation. Recent developments in radar, electro-optical, and inertial sensors will provide a variety of options for implementing the required sensor functions. Many of these developments offer the combined benefits of improved performance and reduced cost. The application of state of the art microcomputer techniques is essential for stand-off weapons. Rapid processing is required for sensor data, guidance processors, and navigation functions. Use of federated computer architectures and standard languages will reduce the cost of the required computational functions.

The purpose of this symposium is to explore the requirements, system trade-offs, and design characteristics involved in stand-off weapon concepts and components and to examine the functions and systems integration issues required to enable effective utilization of tactical precision guided stand-off weapon systems.



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### **ACKNOWLEDGEMENTS/REMERCIEMENTS**

The Panel wishes to express its thanks to the Canadian National Delegates to AGARD for the invitation to hold this meeting in Ottawa and for the facilities and personnel which made the meeting possible.

Le Panel tient à remercier les Délégués Nationaux du Canada près l'AGARD de leur invitation à tenir cette réunion à Ottawa, ainsi que pour les installations et le personnel mis à sa disposition.

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\*Published in classified volume CP 388 (Supplement)

†Not available at time of printing

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## THE UNMANNED AIRCRAFT AS A FORCE MULTIPLIER

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### SUMMARY

A principal role for small military RPVs now and in the foreseeable future is to enhance the effectiveness of other weapon systems, in particular those capable of delivering munitions against distant targets. The requirement is for a 'force multiplier' which can be used to increase weapon effectiveness and weaken the offensive and defensive capability of the enemy.

The intention of this paper is to give the non-specialist an appreciation of how RPV System design and configuration are influenced by a requirement to act as a force multiplier. The approach adopted is to concentrate initially on the target acquisition role and show how mission requirements impinge on every level of RPV system design. The discussion is then generalised to other weapon support roles and finally to the implications of a multiple-role capability on RPV system equipments and operation.

### 1. INTRODUCTION

#### 1.1 Force Multiplication

Force multiplication can be applied in two fundamental areas:-

- (a) The effectiveness of a weapon system can be improved by providing external support to the missions that use the weapon, to weapon aiming and delivery, and to the assessment of mission success.
- (b) The weapon user must strive to make optimum use of the resources at his disposal. The force multiplier can support him by providing information that will help him in deciding what resources to use, where to use them and when to use them.

#### 1.2 The Potential of the RPV System

The attributes of the RPV that make it suitable for use as a force multiplier include the following:-

##### (a) Flexibility

RPV systems can perform a very wide range of mission types. Some are unique to RPVs and others are only achievable by other means at an unacceptable risk to human life.

##### (b) Cost Effectiveness

The cost of adding RPV Systems in support of existant weapon systems is potentially small in comparison to the overall weapon system procurement and operating costs.

##### (c) Adaptability

An RPV system can be dedicated to a particular role or it may perform several role types through the exchange of payloads and ground equipment.

##### (d) Ease of Development

RPV systems have reached a stage of maturity where they can be developed quickly and configured to suit the budgetary constraints and requirements of the user. The range of technology and sophistication applicable to RPVs is possibly wider than for any other category of military equipment.

### 2. THE TARGET ACQUISITION ROLE

#### 2.1 Requirements

The target acquisition role is performed to provide target classification and position data for indirect fire weapon systems, such as artillery or MLRS. Accurate data must be supplied in near real time and, if moving targets are to be engaged, information concerning their speed and direction of travel must be passed to the weapon system. During an engagement the RPV may correct weapon fall of shot and thereafter assess damage.

#### 2.2 Outline Target Acquisition Mission

The following paragraphs provide a simple outline of a target acquisition mission.

(a) Mission Definition

The tasking organisation generates task requirements, such as a route search within a specified area. A mission is a string of such tasks and mission definition entails appending new tasks to current missions, deleting obsolete tasks and defining new missions.

(b) Mission Planning

Mission Planning involves the definition and maintenance of a set of data that is stored in the Ground Control Station and transferred in segments to the RPV. The airborne data comprises a flight control program and RPV sub-system commands, for autonomous control of on-board equipments. The Ground Control Station data is used to monitor and control the progress of the mission.

(c) RPV Launch

The RPV is assembled from a few component modules. It then undergoes a series of pre-flight tests and, if serviceable, is launched.

(d) Transit

The RPV transits in radio silence under autonomous control. At specific points on the mission route communications with the Ground Control Station are established and navigation and mission updates are performed.

(e) Target Acquisition

When a task area is reached, the electro-optic sensor is used for target search, detection and recognition. The RPV may continue to follow the planned route or it may be controlled manually.

(f) Target Location

When a target is recognised it is marked on the sensor display. Its position is determined from current RPV location and height, RPV attitude, sensor pointing angles and local terrain height. Recognisable map features may also be used to locate targets.

(g) Target Reporting

Target reports are made in near real-time using ground communications equipments specified by the user. Where delays must be minimised direct transmission of digital data is desirable.

(h) RPV Recovery

When the last task has been completed the RPV flies an autonomous return leg to the recovery area.

3. A TARGET ACQUISITION RPV SYSTEM3.1 The System

An RPV system configured by Ferranti for roles that include target acquisition is shown in Figure 3.1 and comprises the following equipments:-

- o Thermal Imaging Payload
- o Air Vehicle
- o Ground Control Station
- o Launch Vehicle
- o Secure Data Link System
- o Maintenance Facility
- o Troop Command Post

In this paper the RPV system operational unit will be referred to as a Troop. A Typical Troop might hold two Ground Control Stations, two Launch Vehicles, a Data Link System and 5U RPVs with payloads; and, in addition, user-related vehicles to support Troop command, RPV supply and personnel. Two such Troops might be served by a common Maintenance Facility.

3.2 The Air Vehicle

The Air Vehicle, shown in Figure 3.2, is a small aircraft of simple but robust design, configured for stability as a sensor platform, for ease of assembly and maintenance, and for high survivability.

The Air Vehicle contains a data link terminal and a navigation and flight control unit, complete with control processor to permit autonomous operation.

### 3.3 The Ground Control Station

The Ground Control Station is configured in a container, as shown in Figure 3.3. The Ground Control Station provides facilities for:-

- o compiling tasks into detailed mission plans
- o programming and controlling RPVs via a secure data link
- o updating mission plans while the RPV is in flight
- o reception, display, control and recording of sensor imagery
- o location of targets
- o reporting task results to the user organisation.

### 3.4 The Launch Vehicle

The Launch Vehicle, shown in Figure 3.4, includes a pneumatic launcher, pre-flight test equipment, a data link terminal for short range communications with the RPV, and equipment for RPV recovery.

### 3.5 The Data Link System

The data link transmits commands from the Ground Control Station to the RPV and video and housekeeping data from the RPV to the Ground Control Station. It also controls the RPV immediately after launch and prior to recovery. The principal constraints are to maintain the quality and security of information transmitted while the system is operating in a hostile environment, and to ensure that transmissions do not put the RPV system at risk.

The Data Link System comprises three terminals. The Ground Data Terminal serves the Ground Control Station, but is sited remotely from it as a survivability measure, as shown in Figure 3.1. In addition to transmitting and receiving data, it measures bearing angle and line-of-sight range to the RPV, which are used in the Ground Control Station to determine RPV position. The Air Data Terminal is mounted on the RPV and comprises an electronics unit and an antenna, both located at the rear of the fuselage and replaceable as a complete assembly. The Launch and Recovery Data Terminal contains a subset of the Ground Data Terminal electronics and is used to control the last few km of flight to the recovery point.

### 3.6 The Maintenance Facility

The maintenance facility provides test and repair facilities to support the Troop at field level.

### 3.7 The Troop Command Post

The Troop Command Post is an optional vehicle that can perform a co-ordinating role in large Troops containing, for example, three or more Ground Control Stations. Its functions are covered later.

## 4. SYSTEM DESIGN FEATURES

The aim of this Section is to show that the ability of the RPV system to act as a force multiplier influences every level of system design, using the target acquisition role to fix ideas. Three examples are presented, each covering a different level of system detail. First, command and control requirements for an RPV Troop are considered. Then the Ground Control Station is used as an example of a self-contained equipment, and the reasoning behind its configuration is outlined. Finally, the RPV electro-optic sensor is discussed as a representative sub-system.

### 4.1 System Command and Control

#### 4.1.1. General

Command and Control of the RPV Troop is concerned with technical direction - such as task allocation, and with tactical control - such as deployment.

#### 4.1.2. Technical Direction

The user command structure should contain a single tasking agency responsible for collecting tasks, filtering them, allocating priorities and passing them forward to the RPV Troop.

In a Troop which deploys several Ground Control Stations operating simultaneously, the complexity of task allocation could be relieved, and the efficiency increased, by interposing a co-ordination centre between the tasking agency and the Ground Control Stations. This centre would be part of the Troop and responsible for accepting tasks and allocating them to the most suitable Ground Control Station/RPV combination.

#### 4.1.3. Tactical Control

Tactical Control is largely concerned with deploying Ground Control Stations so that they can maintain coverage of the area of interest. It is probably unreasonable to expect the tasking agency to perform it because it would require detailed and current status information which could overload the agency.

Responsibility for tactical control would be best allocated to the co-ordination centre, which could readily acquire Ground Control Station status data and update itself on the future intentions of the tasking agency.

#### 4.1.4 The Troop Command Post

The co-ordination centre can be implemented in the form of the Troop Command Post shown in Figure 3.1. A single tasking agency at headquarters level collects tasks and passes them to the Troop Command Post, which then distributes them to individual Ground Control Stations. It also maintains a view of the tactical situation and administers the ground equipments accordingly.

In the Ferranti System the Troop Command Post contains a subset of the display, computing and radio communications facilities of the Ground Control Station. This allows a Ground Control Station to perform Troop Command Post functions as a reversionary measure.

#### 4.2 The Ground Control Station

The Ground Control Station is responsible for detailed mission planning, mission execution and task reporting. Mission planning influences how well mission resources are used. The weapon system it supports is dependent on the target information collected during mission execution and transmitted during mission reporting.

##### 4.2.1 Mission Planning Requirements

Each RPV mission is considered as a string of tasks. This concept is central to the design philosophy of the Ground Control Station and, in particular, to mission planning. Some key mission planning factors are as follows:-

###### (a) Mission Duration

RPV missions can last for over 5 hours, and it must be assumed that during this time task requirements could change. It follows that it is not possible to launch an RPV with a complete pre-programmed flight plan and it is sensible to view each task as a separate entity and provide for mission planning to continue in parallel with execution.

Planning must allow tasks to be changed in an orderly, controlled fashion without complex operator actions and thought processes. This flexibility is realised by a mission planning algorithm which fixes tasks in the immediate future, but which may change tasks outside the selected time frame. However, the algorithm must allow manual revision of the plan if necessary.

Operators must keep up to date with changes and, in addition, must be confident that they are in control of the mission. Consequently, an integrated map and data display system is required that can show them not only the entire the mission route but also the vicinity of an operating RPV in detail.

###### (b) Task Priorities

Each task must have a priority relative to the others. The parameters from which priority is deduced is a matter for the operational staffs, but they must be expressed in terms suitable for machine processing.

###### (c) Task Coverage

It is important to relieve the crew of responsibility for defining the detailed track of the RPV in the task area, since this is a time consuming activity. The recommended concept envisages a number of pre-defined flight paths stored in the Ground Control Station, one of which is allocated automatically to the mission plan when the task is allocated. Prior to allocation, the crew may display the planned path and alter any of the parameters - waypoint position, height and so on. They may also select and then modify any of the other stored paths.

This approach ensures that, whatever the loading on the GCS crew, the RPV will adopt a sensible flight plan on arrival in the task area.

###### (d) Task Acceptance

There is no merit in planning tasks which, for reasons of terrain or weather, cannot be performed. Terrain determines the minimum height from which the RPV can communicate with the Ground Data Terminal. Low cloud limits the height at which the RPV may fly if it is to see the ground. Unfavourable combinations of terrain masking and weather will render some tasks impossible.

The system must automatically reject tasks which, because of weather or terrain, are likely to be unproductive. An algorithm which will accept or reject tasks according to a set of simple rules is desirable, backed up by storage and processing facilities for meteorological and terrain height data.

##### 4.2.2 Mission Execution Requirements

###### (a) Sensor Field of View

Variation of ground coverage and image orientation presented to the Ground Control Station operator can lead directly to loss of sense of position and direction.

It must be possible to directly relate the position and track of the RPV, and the area covered by the displayed imagery, to the local terrain. This may be achieved by a moving map display with graphical overlay facilities.

(b) RPV Transit Speed

At typical speed, the RPV can transit a corps frontage in 20 minutes or less. Two considerations emerge. First, the GCS crew must 'stay ahead' of the aircraft (pilots will appreciate this requirement) and second, a ground object will cross the sensor field of view in a matter of seconds, allowing little time for recognition or identification.

Functions relating to the RPV routing must be performed automatically, with facilities for the crew to override or adjust the flight between, and over, tasks. However, manual control should be an exceptional activity. These control features should also be supported by an integrated display of map imagery and flight data.

It must be possible to retain an object in view for prolonged periods for recognition or monitoring. This can be achieved by a mode of sensor control that can track an object or, alternatively, by freezing the display imagery.

(c) The Battlefield Environment

The efficiency of the crew will decline as fatigue and stress build up. Battle damage will diminish Troop resources.

Battle fatigue must not be aggravated by physical and mentally demanding operating procedures. Hence direct operator interaction with the RPV flight control system and the sensor steering mechanism should be prevented. Algorithms should be employed for automatic control and, where manual intervention is required, they should transform simple operator commands into the corresponding actuator demands. This is not a trivial problem, but it need only be solved once at the design stage, rather than during individual operator training on a less sophisticated system.

Battle damage and redeployment are also inevitable, and the design must allow RPV control to be maintained with minimum disruption, by transfer of RPV control between Ground Control Stations.

4.2.3 Mission Reporting Requirements

Whether the Ground Station outputs lists of targets or evaluated assessments is conditioned by operational factors. On the one hand the lists could be very long and the onus of evaluation would be transferred to the RPV system user. On the other hand, evaluation in the Ground Control Station upgrades the crew from observers to intelligence analysts and requires additional facilities. Moreover, engaging the crew in analysis distracts them from observation and may reduce the rate at which tasks are performed.

By whatever means reporting is organised, the method of listing observed objects and their locations into the Ground Control Station data base should be straightforward.

4.2.4 Ground Control Station Design

The Ground Control Station is based around a three-man console, with a layout as shown in Figure 3.3. The three operators are:-

- o A Mission Controller, responsible for compiling tasks into missions, keeping mission plans current, RPV flight control and task reporting
- o A Payload Controller, responsible for handling the mission payload, for the detection of targets, and for their classification and location
- o An Analyst, specialising in detailed examination of imagery in support of the Payload Controller.

Each operator is provided with a TV monitor and a lightpen for interaction with a compact but powerful computer system, which provides easy to use menu selection facilities for mission planning and execution. The Payload Controller's and Analyst's monitors display real time video imagery, and the light pens are used to mark and describe targets. The Analyst has a frame freeze option for extended observation of targets. The Mission Controller and the Payload Controller have joysticks for RPV control and sensor steering respectively.

Between the Mission Controller and the Payload Controller workstations shown in Figure 4.1, is a Ferranti Combined Map and Electronics Display (COMED), which mixes high resolution map imagery with computer alphanumerics and graphics, as shown in Figure 4.2. This provides a focal point for mission planning and execution, since it can display the entire mission route or the area around the air vehicle at a considerably larger scale. At this scale the map moves in response to RPV motion, the planned route is shown and in task areas the sensor footprint on the ground is drawn to scale. This facility, combined with the display of natural looking horizon-up imagery, allows the ground features and targets to be related on the map, aiding both navigation and target location.

Facilities also include a digital terrain data base, ground communications equipment and environmental protection.

The computer system executes the following software functions, which reflect the operational functions performed by the Ground Control Station:-

(a) Mission Definition

This function compiles incoming tasks into missions. It originates new missions, changes tasks in existing missions (according to defined rules), allows manual revision of missions and maintains

mission and task data for display.

(b) Mission Planning

This function converts current task information into a detailed mission description. It generates the mission route between tasks, taking into account terrain height, data link intervisibility requirements, cloud level and areas to be avoided, for reasons such as survivability. In addition, it generates a default route to cover a task which can be adopted if no time exists for the operators to improve on it, and displays the current route on COMED at any time during a mission.

The mission data is used to generate and maintain a flight program that is transmitted via the secure data link for execution by the RPV control processor.

(c) Mission Execution

This function contains a number of sub-tasks which include:-

- o monitoring and display of RPV and mission status
- o a simple interface for operator control of the sensor and RPV
- o computation of RPV position for navigation updates, including map transformations
- o target location from sensor pointing angles, RPV attitudes and RPV and Ground Data Terminal positions
- o control of the Ground Data Terminal control.

(d) Mission Reporting

This function displays on COMED objects marked by the operators using their light pens. It compiles lists of these objects and condensed reports, then formats them for onward transmission.

4.3 The Electro-optic Sensor

The electro-optic sensor is fundamental to the target location role. Its performance determines the target types that can be acquired, the range from which they can be recognised and the accuracy with which they can be located. It must also comply with design constraints imposed by its operation in the RPV.

4.3.1 Requirements

The performance requirements for the electro-optic sensor are:-

(a) Twenty Four Hour Operation

If day and night operation is required and sensor performance is to remain reasonably consistent throughout, then a thermal imager operating in the 8 - 14  $\mu$ m region is required.

(b) Ground Coverage

Flexibility of operation and survivability can be enhanced by providing observation from a stand-off position along a simple and easily held RPV flightpath. Consequently, the sensor should be steerable over the lower hemisphere beneath the RPV.

The imagery presented to the Ground Control Station operators must appear natural if disorientation is to be prevented. Human factors experiments have shown that horizon-up imagery is best where a look-round capability is required. This can be readily achieved by mounting the sensor in a 'pan and tilt' arrangement beneath the air vehicle i.e. with degrees of freedom in azimuth and elevation.

(c) Target Detection and Recognition

The sensor must meet the conflicting requirements of providing wide area coverage for target search and detection and a close up view for recognition. This can be achieved by providing a switched or zoom field capability.

In addition, target recognition requires a steady image. Therefore the sensor must be stabilised against the flight environment of the RPV.

(d) Target Location

The sensor pointing angles must be measured accurately for the computation of target location relative to the RPV.

(e) Size, Height and Cost

Sensor design and construction must be constrained within strict weight and dimensional limits. The cost of the RPV and its payload must be kept low since, by virtue of the roles it is tasked to do, survivability is inherently lower than for manned aircraft.

(f) Flexibility

There is a great temptation, given the above constraints, to limit the sensor specification to one particular requirement. However, provided flexibility is considered from the outset, no major

penalties need be incurred in allowing a simple upgrade path to better performance or supplementary roles.

(g) Interchangeability

The RPV is likely to be launched from a site close to the FEBA. This precludes sensor repair at the Launch Vehicle and ease of payload exchange must be a principal design aim.

4.3.2 Electro-optic Sensor Design

The Ferranti electro-optic sensor package is shown in Figure 4.3 and comprises a thermal imager mounted on pointing and stabilisation gimbals and housed in a small protective ball.

The thermal imager is a compact design, suited to RPV applications. It is fitted with an equally compact telescope with a  $16^\circ \times 12^\circ$  field of view for target detection and a  $4^\circ \times 3^\circ$  field of view for recognition.

The imager is mounted on a four-axis gimbal system. Two inner gimbals are gyro stabilised to give a residual sightline jitter less than 60  $\mu$ rads. The outer gimbals provide steering over  $+-220^\circ$  in azimuth and from  $+20^\circ$  to  $-120^\circ$  in elevation. The sightline can be slewed at up to 1 rad/s to allow the operator to demand reasonable slew rates in the presence of typical RPV body rotations.

The gimbal structure is based on a single-armed cantilever arrangement which offers simple assembly, ease of machining and ease of access to the imager and telescope. It also offers flexibility in being able to accommodate a range of sensor options including:-

- alternative thermal imagers
- an alternative daylight TV and LLTV package
- a thermal imager and laser rangefinder and/or designator

The sightline direction is held fixed in inertial space by stabilisation gyros and is measured with respect to the airframe to an accuracy of 3 mrad.

The gimbals are housed in a protective ball, which provides a controlled environment and supports and slaves a small germanium window to the thermal imager sightline. Low radar cross-section and high survivability have been taken into account in the ball and window design, and the ball is sectioned for easy access to the interior.

The complete payload is located in the RPV fuselage by a rigid mounting plate that can accommodate the inertial sensor unit of the navigation sub-system, to provide close coupling between sensor pointing angles and measured RPV attitude.

Value engineering has been applied to the design, resulting in the replacement of high value components found in sensors for manned aircraft e.g. high technology angle transducers and gas couplings, by lower cost alternatives. Lessons learned in this process have been applied to an even greater extent in the daylight TV payload shown in Figure 4.4.

5. WEAPON SUPPORT ROLES

In this short section other RPV weapon support roles are defined and RPV equipment characteristics determined by role type are discussed.

5.1 Weapon Support Roles

In Section 1.1 it was stated that a force multiplier can influence both weapon system performance and user efficiency. A selection of RPV roles, with their applicability to these two aspects of force multiplication, is listed below.

Laser Designation	Weapon completely dependent on RPV
Target Acquisition	Weapon performance significantly improved by RPV data
Electronic Countermeasures	Mission success increased Threat to weapon system reduced
Radar/Comms Harrassment	Mission success increased Threat to weapon system reduced
Decoy	Mission success increased Threat to weapon system reduced
Surveillance	User deploys weapon system more efficiently on the basis of RPV data
Electronic Support Measures	User deploys weapon system more efficiently on the basis of RPV data.

It is impossible to discuss the above roles individually in this paper. However, an appreciation of the consequences of role type on RPV system design can be given by discussing a few key factors.

## 5.2 Recoverable vs Disposable RPVs

Some roles, such as photographic reconnaissance, require RPV recovery. Others, such as an anti-radar attack drone, require a disposable RPV by definition.

If recovery is an inherent requirement it is sensible to design the air vehicle to be re-usable. If the role demands expendability then minimising airborne hardware costs is a major design consideration. Since disposable RPVs will be procured in considerably greater numbers than ground equipment, it is especially worthwhile to trade cost savings in the former against cost and sophistication in the latter, because this could significantly enhance overall system effectiveness.

Other arguments for recovery tend not to be based on role considerations directly, but on trade-off between the cost of the mission payload and the increase in life-cycle cost attributable to the recovery facility.

## 5.3 Autonomy of Operation

The degree to which the RPV can function autonomously depends entirely on one factor - whether the role demands that data be transferred to it from the ground, or from it to the ground, or both.

Given that a link is required, incidental overheads will include terminals in the RPV and on the ground each with associated processors and possibly encryption devices. Some form of manned Ground Control Station will be required, and since it is unlikely that these can be afforded or manned in large numbers, or have a capability to control more than one or two RPVs simultaneously, the task performance rate will be relatively low. On the other hand, the link can enhance effectiveness since it can be used to improve navigation accuracy, flight control and recovery.

The truly autonomous RPV is a fire and forget vehicle. It is pre-programmed before flight, and in consequence, the Ground Control Station needs only to transfer the mission plan to it once. This task can be performed quickly for a large number of vehicles, and hence a high task rate could be expected. Thus the RPV, the Ground Control Station and the whole of system operation are considerably simplified.

Under these circumstances a single ground equipment may perform both the functions of the Troop Command Post, described in Section 4.1 and those Ground Control Station functions associated with verification of task data, mission route planning and the computation and transmission of launch and navigation data. Tasks would be received and compiled into complete mission plans for allocation to the Troop launch sites.

## 5.4 Quality vs Quantity of Task Coverage

The fundamental requirement for some roles is quality of performance, and these roles will be described as 'specific'. For example, a laser-designation task requires the RPV to loiter and illuminate a particular target at the time of weapon delivery and success is dependent on the performance of an individual RPV. On the other hand, the success of an electronic countermeasures role may depend entirely on the cumulative effect of several RPV tasks covering a wide area. In this case the success criterion is the amount of coverage, although obviously each RPV must perform adequately. Roles where quantity is the dominant factor will be described as 'generic'.

'Specific' roles often demand a level of performance from the payload achievable only using advanced and expensive technology and thus require a recoverable RPV for cost-effective operation. If the role also demands data transfer to the ground then the mission payload can dominate the design of the entire RPV system. It then becomes cost effective to provide for future role extensions. For example, in Section 4.3 the considerable effort applied to the electro-optic sensor design in the target acquisition role was addressed. Under these circumstances it made sense to include, from the outset, flexibility to accommodate related roles such as laser designation. This flexibility was attained without significant cost penalty.

A 'generic' role may require RPVs to be brought into use quickly and in large numbers. Multiple launch from a single ground vehicle, with minimum operator involvement, is a key factor and the problems associated with the recovery of large numbers of low-cost vehicles makes expendability attractive. Deployment in large numbers poses control and data handling problems which could saturate available ground equipment. Therefore missions should be largely, if not totally, autonomous.

Individual RPV loss is not critical to overall effectiveness in a 'generic' role, hence expenditure on achieving performance and survivability can be traded off against RPV cost and numbers procured.

## 5.5 Multi-role vs Single Role Options

Two extreme approaches are a single RPV system that can meet all the role requirements or, alternatively, the deployment of a range of RPV systems, each dedicated to a single role.

A single RPV system capable of performing all the roles described in Section 5.1 is unlikely for three reasons. First, the disparity of roles is such that it is unlikely that all could be met economically by a single airframe and launch system. Second, the ground control equipment and manpower would have to be adaptable to all the roles required, with obvious implications on complexity, operator skills and logistic support. Third, communications, command and control arrangements necessary to serve a variety of tasking agencies would be difficult to implement during air/land battle. These conditions are exacerbated if the RPV is required to perform several roles in the course of a single mission, and by the need to resolve contention for its services at command level.

An alternative policy is the deployment of a variety of RPVs, each optimised towards a single role or configuration, and supported by dedicated ground equipment. This is technically feasible and operationally attractive in that it could place RPV support under the direct control of those who would benefit from it. However, it is unlikely to be affordable, both in money and manpower. Moreover it will lead to problems with logistics, organisation, communications and training.

A compromise between the above options involves employing a small number of RPV systems, each configured for a limited set of roles and designed to comply with the following constraints:-

- o compatibility with the user command and operations structure
- o maximum commonality of equipment
- o maximum commonality of operating and support procedures.

## 5.6 Design for Multi-role Usage

User organisations vary in size, in their role requirements and the weapon systems at their disposal, hence the definition of a unique combination of RPV systems with universal application is an ideal that is unlikely to be realised.

The aim in practice, therefore, is to achieve effectiveness in as many roles as possible using the minimum number of RPV system variants. The categories described in Section 5.2 revealed a tendency towards two basic system types. The first deploys a recoverable RPV, transmits data to the ground and performs 'specific' roles. The other deploys expendable RPVs in large numbers, each performing autonomously. These systems form useful examples on which to base comments on how a multiple-role requirement can affect the RPV system and its constituent equipments.

### 5.6.1 The RPV

Some general points are as follows:-

#### (a) Payload

Where the role requires the RPV to be expendable, the mission payload should be cheap, effective and reliable. Otherwise, ways of achieving the same result with a recoverable RPV or different means entirely should be considered.

Where the role requires data-link communications with the ground, payload design is constrained by the need to interface with the Air Data Terminal and by the fact this terminal will take up valuable weight and space.

Where the role is 'specific', mission payload performance may dominate total system design. Having achieved that performance it is then worth designing for flexibility to accommodate related roles or future role enhancements.

Mission payloads should fit more than one RPV, since a new requirement e.g. for extended range, could be met by the same payload in a different airframe. It is tempting to think of role changes in manned aircraft and as a consequence effect RPV mission role changes by replacement of payloads at the launch site. However, the influence of role change propagates throughout the system and co-ordination of units within the Troop and at higher command levels becomes the main problem.

#### (b) Airframe and Avionics

On paper the RPV has the straightforward task of flying to the operational area and providing a stable platform for the payload. For this reason air vehicle design tends to be under estimated. Over fifty performance criteria were explicitly addressed in the design of the Ferranti RPV, despite its deceptively conventional appearance, including several to ensure that this vehicle meets all the requirements for ease of role change.

Having said that, the airframe should not be allowed to dictate system characteristics. In general, the basic airframe cost will be significantly less than that of the payload. Therefore, provided that the number of airframe types is kept comfortably within the ground handling, logistics and maintenance capability of the user there should be no need to compromise mission effectiveness by rigidly adhering to a single airframe.

However, on-board avionics for control of the airframe are expensive to procure and support, so maximum commonality should be sought in this area.

#### (c) Ground Handling Equipment

Variety of RPV ground handling equipment should be minimised. Two basic launch vehicle types would probably suffice for a single user - one for multiple launches of expendable RPVs and one for single launches of recoverable or expendable RPVs. For example, the Launch Vehicle in Figure 3.4 lies in the first category and can be readily adapted to accommodate a range of take-off weights and launch speed requirements.

As far as possible the differences between role types should not introduce fundamental changes to ground handling procedures. Pre-flight procedures should be common to role types and testing should be automated. Maintenance should be limited to replacement of entire RPVs or large modules capable of being exchanged in adverse weather conditions and by operators in protective clothing.

5.6.2 Data Link

Data Link interoperability between RPV systems should be a minimum requirement, with commonality of equipment highly desirable.

5.6.3 Ground Control Station

Several of the activities carried out in the Ground Control Station are common to other roles. These are concerned with the verification of task data, mission route planning and the computation and transmission of launch and navigation data. Other activities are specific to the role.

Ground Control Station design should differentiate between mission dependent features and those that are common between roles. As an example, the integrated map and processing facility based on the Ferranti COMED can display mission planning data and tactical information. In addition, it can be customised to handle the display of mission specific data, such as display of the sensor footprint in the target acquisition role, thereby reducing additional facilities required for role changes. The menu driven software, which effectively leads the operator through sequences of actions, provides a common basis for merging role dependent and role independent procedures.

5.6.4 Troop Command Post

If a system has to perform several roles the need for a Troop Command Post is increased in order to present a single contact point with the operational commanders and to sort and allocate tasks. Common display and processing facilities with the Ground Control Station allows tasks to be compiled into outline mission plans in the Troop Command Post. Assuming that digital communications channels exist between Troop ground units, outline plans can then be transferred from the Troop Command Post processor to the Ground Control Station processor in a format that can be used directly for mission planning. Fast and efficient transfer of high level mission and task information provides a powerful basis for co-ordinated action within the Troop. In addition common facilities permit reversionary operation of a Ground Control Station as a Troop Command Post.

Where an autonomous and expendable RPV system is being tasked, the same display and processing facilities can be used for preparing complete mission plans, again making use of partitioned mission dependent and mission independent software.

6. SYSTEM MANAGEMENT FOR WEAPON SUPPORT

The aim of this paper has been to give an appreciation of how RPV system design and configuration are influenced by a requirement to act as a force multiplier. This resulted in an introspective look at RPV systems. However, the principal challenge for the immediate future does not lie with RPV performance, but in providing the user organisations with facilities to manage a range of RPV systems and roles. Factors that must be considered include:-

- The command structure for tasking RPV systems
- Connecting the RPV systems with the organisation for which it is currently working
- Organisation to cope with role change in battlefield conditions
- Merging the task results from several RPVs to increase overall effectiveness
- Provision of reversionary measures to minimise the effect of equipment losses
- Co-ordination of RPV activities with other organisations - e.g. RPVs cannot be allowed to prevent a hazard to manned aircraft.

While it is beyond the scope of this paper to cover the higher level implications of RPV system management, it is acknowledged that RPV system design and configuration must offer the inherent flexibility to adapt to the user organisation. Section 4 described integrated processing and display facilities which, in effect, replace the arrays of plotters and controls used previously and which form the centrepiece of compact but effective ground stations. However, they were also designed to provide the user of multi-role RPV systems with flexibility to manage those systems to serve his needs. The basis for this flexibility is:-

- Integrated processing and display facilities that can be applied to all levels of RPV system management
- Real time map and graphical display
- Common menu based operating procedures
- Partition of facilities and operator functions common to several roles from role specific facilities and functions.
- Processor to processor data communications.

These features are relevant not only to the RPV Troop but may form basic building blocks for use by tasking organisations and command centres to process and display information from a wide range of sources then transmit it in a form most meaningful to the recipient.

7. CONCLUSION

Force multiplication can be realised by calling on different RPV roles, either singly or in combination. For these roles, and for the target acquisition role in particular, it has been shown that every aspect of RPV system design is affected by the weapon support requirement. Therefore, a multi-role capability requires more than a change of mission payload, and a single RPV system configuration will not be able to cope with every role. On the other hand, to prevent the proliferation of role-specific RPV systems, common features must be identified and used in a limited range of interoperable systems. Such features include mission payloads that can be used in a variety of RPV types, commonality of air vehicle handling and support equipment and procedures, and the partition of mission planning and execution facilities into role-dependent and role-independent categories.

While the RPV system designer cannot solve the problems of managing a multi-role system without entering a partnership with the end user, he can provide a set of basis tools that will allow this partnership to manage RPV systems to obtain maximum gain from force multiplication.

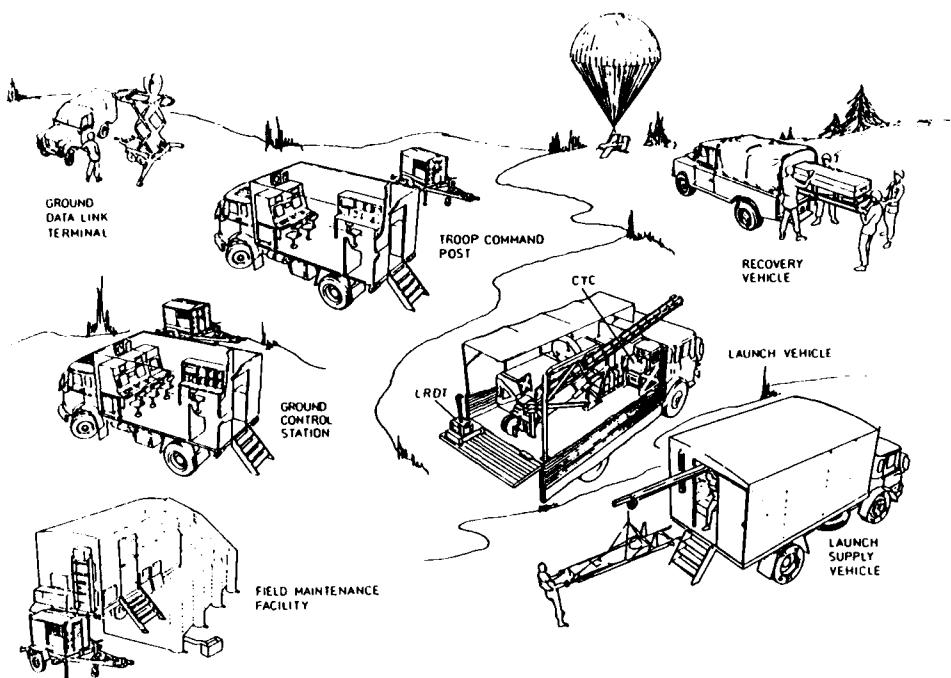


FIGURE 3.1 THE FERRANTI RPV SYSTEM

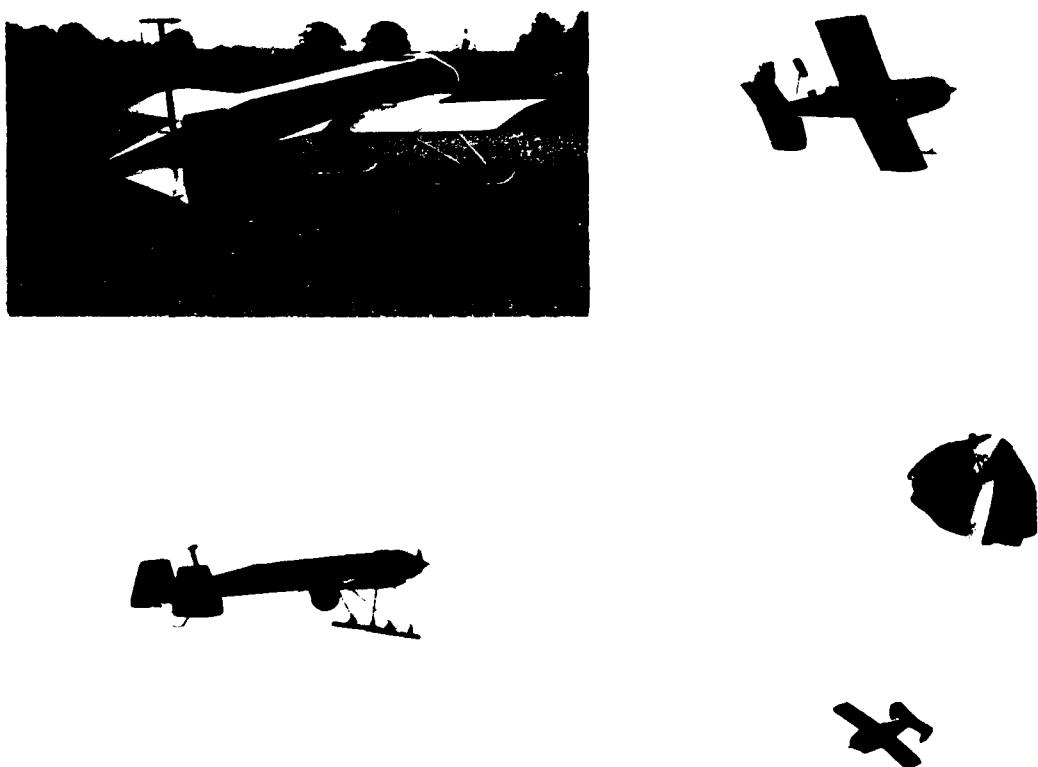


FIGURE 3.2 THE REMOTELY PILOTED VEHICLE

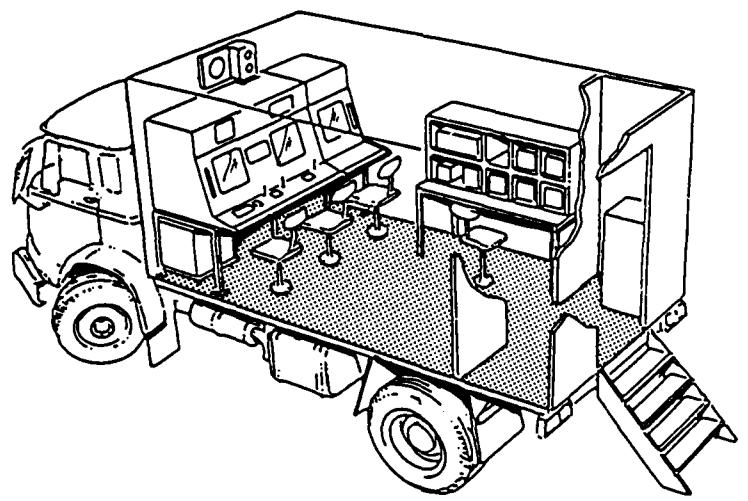


FIGURE 3.3 THE GROUND CONTROL STATION

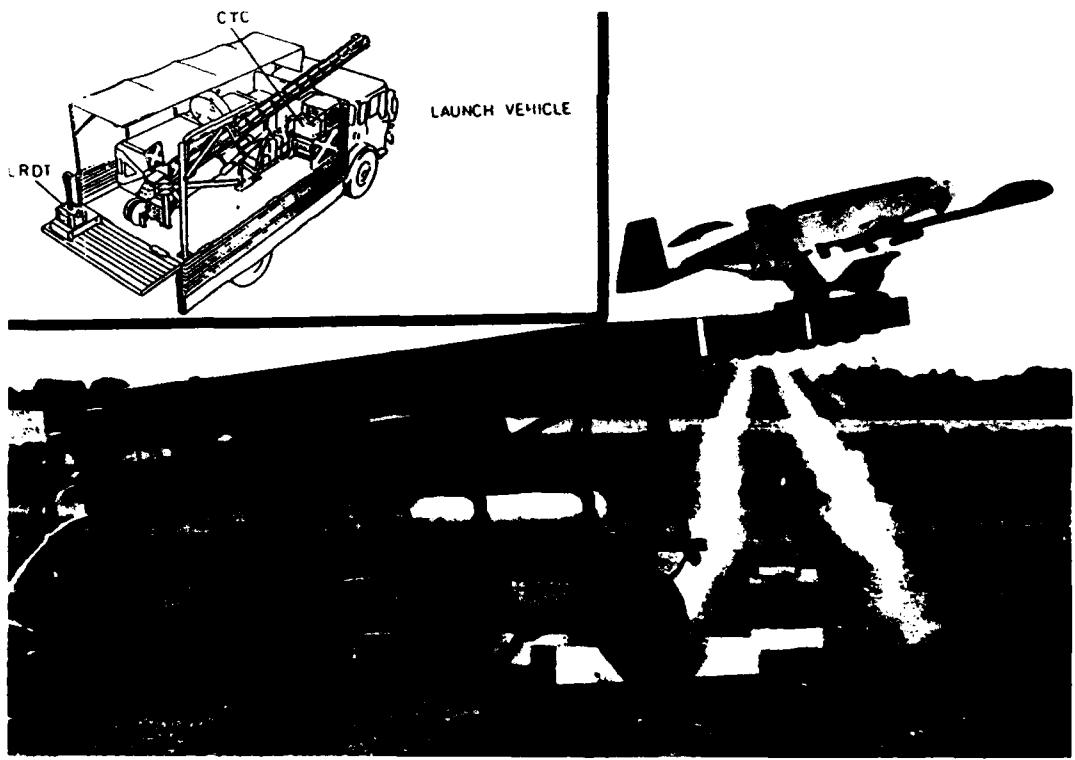


FIGURE 3.4 THE LAUNCH VEHICLE AND PNEUMATIC LAUNCHER

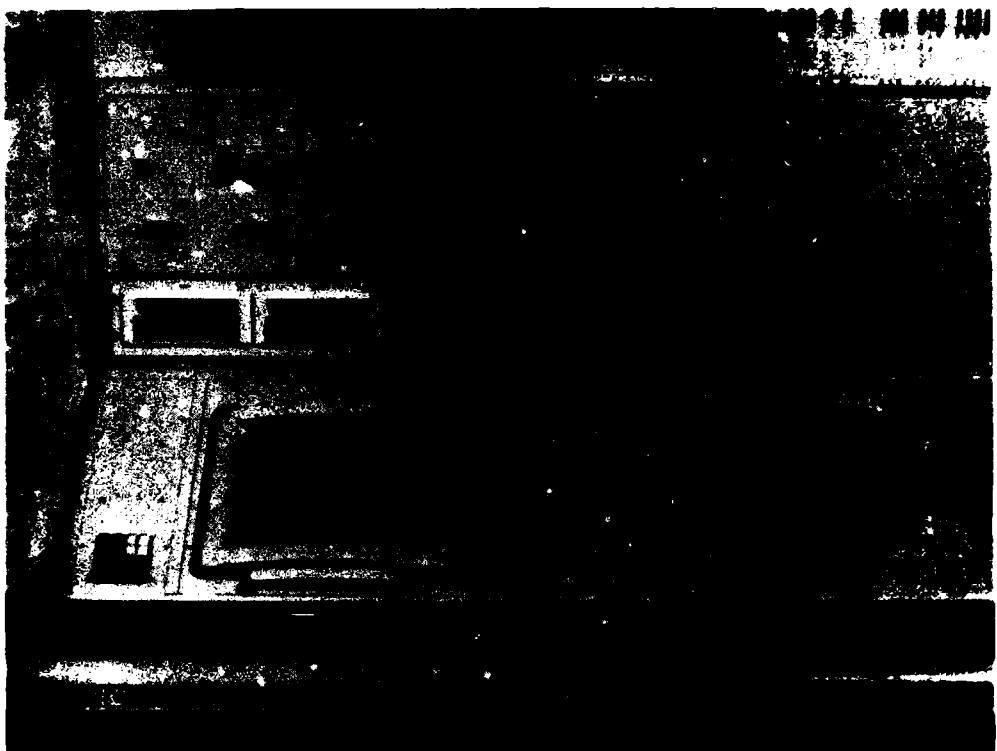


FIGURE 4.1 OPERATOR WORKSTATIONS

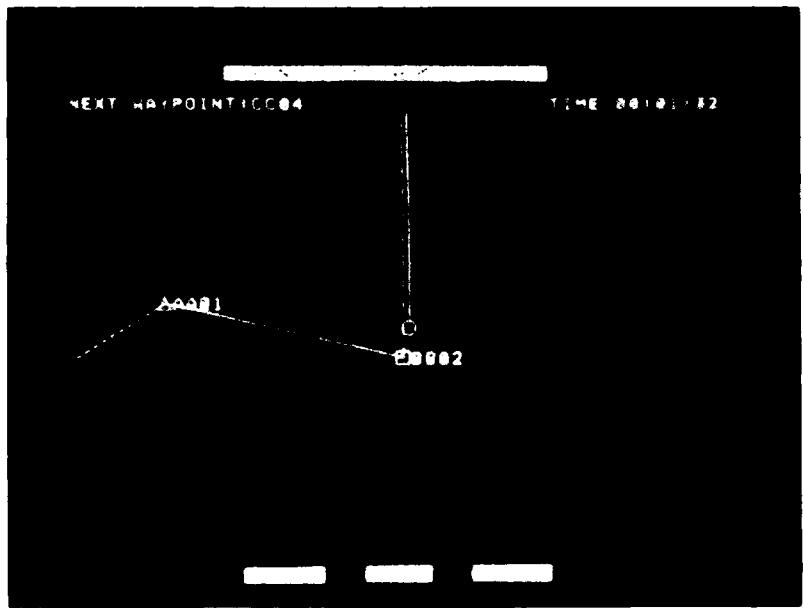


FIGURE 4.2 COMED DISPLAY

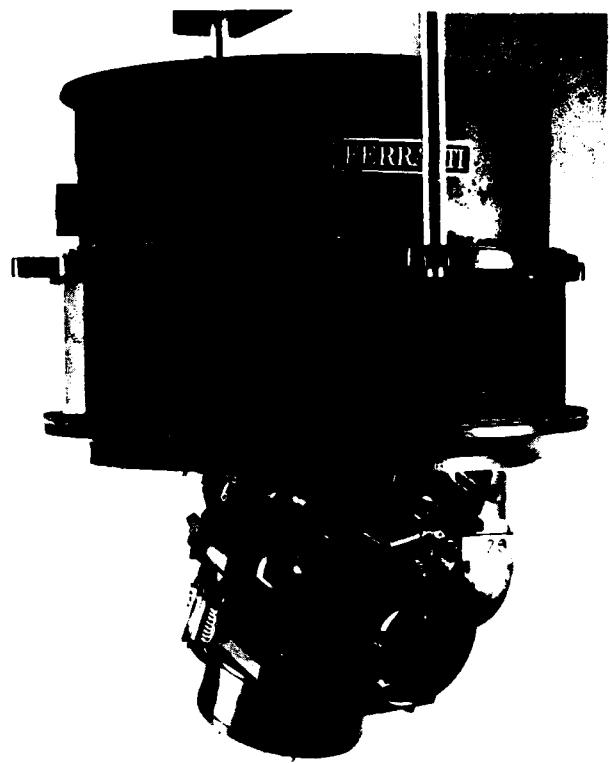


FIGURE 4.3 FERRANTI SENSOR BALL

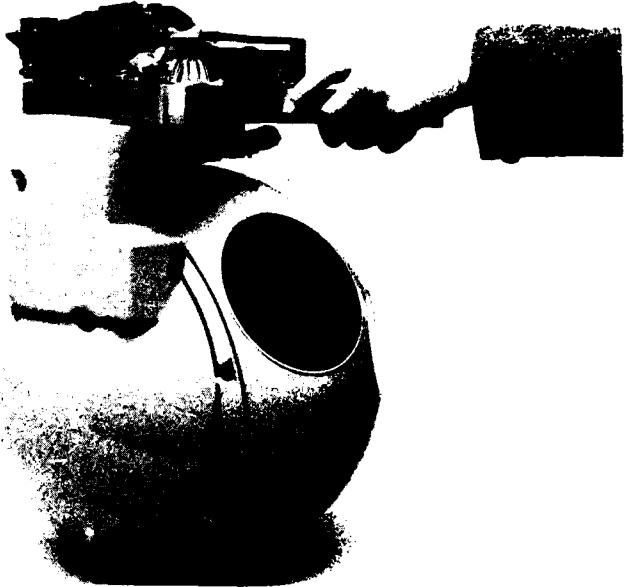


FIGURE 4.4 DAYLIGHT TV PAYLOAD

## NAVY TECHNOLOGY REQUIREMENTS FOR UNMANNED AIRBORNE VEHICLES

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 APC-202

## BACKGROUND AND INTRODUCTION

Historically the Navy use of Remotely Piloted Vehicles (RPVs) has had many pitfalls. Along with the other services, the Navy has applied unmanned air vehicle technology most consistently in the area of aerial targets. Aerial targets are unmanned vehicles developed to simulate threat aircraft and missiles characteristics for engagement by air-to-air and surface-to-air weapon systems. Perhaps as stated in William Wagner's "Lightning Bugs and Other Reconnaissance Drones", there has been an institutional bias toward piloted aircraft. A more likely deterrent could well be the Navy's unique requirement for mobile, at-sea basing of RPVs and the resultant problem of shipboard launch, recovery and maintenance. There is only so much room on-board Navy ships for personnel and equipment, and in many cases the addition of new equipment requires the removal of an existing system. This zero sum game is as applicable to large aircraft carriers as it is to small combatants.

Whatever the reasons of the past for sporadic commitment to RPVs, recent demonstrations of RPV's effectiveness by the Israelis, coupled with advancing technologies has led to a renewed interest by today's Navy and Marine Corps in the use of RPVs. However, due to lack of Navy operational experience with RPVs, initial mission requirements will be satisfied using off-the-shelf industry designs. These baseline systems will provide the means through which Navy personnel could obtain hands-on experience with the tactical applications of RPVs. Revised requirements and updated systems will then be introduced based on: shipboard handling experience, tactics developed, operational scenarios established, and performance requirements identified using the baseline system(s). The updated RPV systems will utilize advanced technology to reduce cost, increase survivability, and improve performance relative to the baseline systems.

Table 1 summarizes the broad range of Navy missions in which RPVs could be used either as replacements of manned aircraft, or to supplement/augment manned aircraft. At the present time, the Navy is addressing: Battlefield surveillance, reconnaissance, and area surveillance.

One specific program is the near term fielding of a short range RPV to perform tactical intelligence data gathering, battlefield surveillance in support of ground troops, and Naval and artillery gunfire support. Representative fixed and rotary wing air vehicle concepts are being reviewed in the context of the total RPV system, including: ground control station, data link capabilities, airborne sensors and avionics, and ground support equipment/personnel and facilities requirements.

Another ongoing program is the Tactical Reconnaissance RPV Program. The mission of this RPV is to complement the reconnaissance role of the F-14 in high threat areas. The baseline system under review/development is a modified BQM-74C aerial target.

Longer term programs include joint Navy Marine Corps efforts as well as tri-service developments with the focus on lowest Life Cycle Cost (LCC) and mission effectiveness approach in the context of the entire RPV system.

## NAVY UNIQUE CONSIDERATIONS

To differentiate the Navy RPV requirements from those of the Army and Air Force, it is necessary to review the fundamental difference of mobile, at-sea basing. While there are some differences in the missions to be performed, most notably the ASW mission, the general requirements for techniques and devices associated with mission execution are on a parity with those of the other services. Therefore, the remainder of this paper focuses on the shipboard environment and its impact on Navy RPV technology needs.

First and foremost is the limited availability of space. This includes deck space for launch, recovery and post recovery decontamination facilities, hanger space for maintenance, below-deck space for the Ground Control Station (GCS), and crew quarters for the RPV system maintenance personnel and operators. Of course, the specific space limitations vary depending on the size of the ship. In general, however, it is a zero sum game insofar as the introduction of the RPV system into the ship requires the removal of existing equipment, facilities and/or personnel. On some smaller surface combatants, vertical (ship) C.G. is sensitive to the point that mast-mounted antennas for RPV command, control and communications may pose a serious problem. Another potential problem on board the smaller and mid-size ships is the necessary use of ladders and hatchways to transport the RPV from below deck to the weather deck for launch. On the larger capital ships and aircraft carriers, the RPV must be integrated functionally and physically with existing air assets. The impact of limited hanger space results in the need to tailor Organizational and Intermediate level maintenance concepts to maximize commonality. The impact of limited crew quarters implies that the functions associated with RPV system operations, including launch, recovery, remote piloting, and data analysis, as well as maintenance will have to be performed on collateral duty basis by the existing crew complement.

The environment unique to extended at-sea operations also raises the need for unique design concepts and technology applications. The salt air environment is extremely corrosive and the RPV vehicle and its internal subsystems as well as ship-mounted launch and recovery equipment must be protected or made from impervious materials. The use of fresh water to decontaminate the RPV is not an acceptable normal procedure due to limited water supplies and the need for specially configured wash-down and drying facilities. This constraint is particularly applicable to recovery concepts which require the RPV to land on the water for subsequent recovery by helicopter or a small boat.

The need for on demand launch and subsequent recovery from a moving ship further introduces technological challenges. Although the orientation of the ship relative to the prevailing winds is somewhat under the discretion of the ship captain, the motion of the deck (roll, pitch and heave) is totally dependent on the weather and sea state. Therefore, there is a need for a launch and recovery capabilities which do not limit normal/safe operations to day light and calm weather conditions. The presence of high near-field air turbulence generated by the ship and its superstructure further complicates the design of the flight control and landing systems.

Another factor unique to at-sea basing is that the ship presents a moving base of operations. Namely, the longer the RPV mission endurance is, the more difficult it is to preprogram the RPV computer with the expected position of the recovery ship. This problem is further complicated during special missions where EMCON conditions prevail. This is one area where the use of artificial intelligence technology may prove beneficial.

Navy operations of RPVs from on-board ships can be in support of either sea operations or land operations. In either case, given that the sensors payload and associated avionics can be interchanged within a single air vehicle, the limited horizon available to a shipboard GCS introduces the need for clever data link schemes. For expendable vehicles, which may be a required compromise for the smaller size ships, a data link relay platform may be required. This will introduce the need for additional assets into an already space-limited situation. This is one area where reliance on space-based satellites may introduce a technology challenge. Another unique considerations is IFF. It is necessary for the ship to be able to positively identify the RPV and distinguish it from anti-ship missiles.

Up to this point, we have addressed some of the unique aspects of the operating environment within which Navy RPVs will be expected to operate. It must be recognized that these operations must be safe and effective during both peacetime conditions as well as during hostilities. It is further necessary to point out that the Navy and Marine Corps operate world wide. Therefore, it can be reasonably expected that a need will arise to deploy RPVs in a part of the world in which the RPV remote operators have no previous experience or familiarity with the local topography, nor up to date information of hostile installations, such as SAM sites.

#### TECHNOLOGY NEEDS AND IMPLICATIONS

Given the shipboard operating environment described thus far, we now turn our focus to selected technology topics which, in my own view, need additional development and design applications before RPVs become a routine part of Navy operations.

For the purposes of this paper, the family of RPVs can be grouped into three classes of short, mid-, and long range vehicles. Furthermore, the Navy's fleet can also be grouped into three classes, namely small combatants, mid-size/capital ships, and large ships which include aircraft carriers. Table 2 presents a possible basing concept for the three classes of RPVs in the context of three classes of Navy ships. The purpose of this approach is to enable the discussion of technology needs from the shipboard basing point of view. It was stated earlier that the technology needs from the mission applications point of view do not vary significantly from those of other services. Discussed first are some design, and therefore technology drivers unique to each of the three classes of ships, followed by a discussion of technology needs common to the three classes of RPVs.

#### Small Combatants

This class of ships, with their severely limited space provisions, impose the most stringent launch, recovery and shipboard maintainability constraints. Therefore, it is assumed that only short range, possibly expendable RPVs could be based on this class of ships.

The lack of adequate deck space for launching rails of VTOL operations would mandate a canister-type launch. For beyond Line Of Sight (LOS) operations, several RPVs may have to be operated simultaneously to provide for real time data and telemetry relay. This mode of operations impacts the cost of the entire system (many vs few), as well as the GCS design and the number of remote operators. A relay form of operations may also be required for expendable RPVs. Of course, the deployment of low cost expendable RPVs would alleviate the shipboard recovery problem.

The general small size of this class of RPVs would probably result in dedicated/single mission vehicles with shorter range/endurance capabilities. The latter should be reflected in a lower cost propulsion system. However, clever design schemes need to be developed to maximize modularity and thus increase mission flexibility and reduce system LCC. For example, modular airframe concepts would allow easy and quick vehicle reconfiguration. The use of light weight material such as plastics, composites and styrofoam would ease shipboard handling and weight constraints. A MIL-STD-1553 data bus architecture, using common data processors and navigation computers would also facilitate sensors and avionics reconfiguration in the form of WRA and SRA (Weapon/Shop Replacement Assembly).

#### Capital Ships

This class of ships relax somewhat the space limitations associated with the smaller ships. However, the mission of this class of ships introduces the need for greater RPV mission flexibility, longer mission radii, and the potential need to operate in conjunction with/in support of manned vehicles. These, in turn, imply a more capable propulsion and flight control/navigation systems to support higher speed operations at the low-to-mid altitude region. The somewhat larger size of this class of RPVs could afford greater reliance on multi-mission capabilities through the use of larger sensors and avionics payload. An over the horizon secure/LPI broad band data link is likely to be required to support the potential broad mix of onboard sensors, including radar, IR and video. Limited data processing may be provided within the RPVs avionics payload.

Launch concepts from this class of ships could include vertical takeoff or rail launch. Recovery concepts could include vertical landing using rotary wings, parachutes/parafoils, or fabric wings. While rotary wing technology appears to be the most desireable concept due to the availability of continuous control during approach, there is a need to examine the tradeoff with lower cruise speed and higher maintenance typically associated with this class of VTOL aircraft.

#### Large Ships

RPVs deployed from this class of ships are likely to approach the size of piloted aircraft. It is reasonable to expect that this class of RPVs will present the high cost end of the family of RPVs, but would also provide greater mission flexibility as well as some unique mission capabilities not possible (cost and safety) with manned vehicles. Some of these unique capabilities could include extremely long mission endurance, possibly measured in days or weeks, extremely maneuverable platforms, or very high altitude surveillance platforms. This class of RPVs is likely to demand maintenance, logistics and operating facilities/personnel not much different from those required to support carrier based aircraft. It can also be expected that the lower cost of the airframe, relative to manned aircraft, will be offset by the higher cost of the specialized sensors and avionics payload.

Launch and recovery of this class of RPVs will be similar to manned aircraft. The ability to use catapult and arresting gear to sustain the typical high tempo carrier air operations should be a design consideration. The use of single point pressure refueling and automatic wing fold mechanism are two more ship suitability considerations.

#### Common Technology Needs

Shipboard recovery- This technology area is both unique to the Navy and the one in greatest need for technology advancements. It is necessary to identify and examine viable recovery alternatives, as well as a practical means of evaluating competing concepts. Some possible technology areas include: VTOL-capable RPVs, vertical landing concepts using parachutes/parafoils, or in-flight deployable fabric wings. It is also necessary to study and understand the impact on the ship posed by the recovery schemes. The use of ship-mounted recovery equipment, such as nets, hooks or even energy absorbers, as well as the potential need for special ship maneuvers must be carefully assessed. And, in all of these cases, a reliable methodology must be formulated to evaluate the cost tradeoffs between the various technologies and concepts. An RPV-oriented costing model needs to be developed to enable the Navy to conduct technical evaluations of expendable concepts, single-mission concepts, and multimission-capable concepts. This cost model must include not just the air vehicle, but also the rest of the RPV system elements, including: GCS, maintenance and support facilities, data reduction, sensors and avionics, software, air vehicle subsystems, and personnel. Only a comprehensive cost model can and should be used to identify RPV system cost drivers, and therefore the specific technologies which can contribute to LCC reduction. For example, if the sensors and avionics part of the system make up more than half of the LCC, while the air vehicle is only 10% of the cost, avionics and sensor-related technologies should be focused on.

Configuration technology- This is another area where great strides towards lower LCC can be made. The adaptation of modularity (erector set) concept to both the airframe and to the sensors and avionics would increase commonality, reduce turn around time, increase mission flexibility, and provide for greater optimization of RPV system configuration to changing circumstances.

The concept of modularity is also applicable to the GCS, and should take into account the need for rapid reconfiguration and transportability from ship to shore in support of amphibious operations.

Manufacturing technology- It is mandatory that the cost of RPV manufacturing be significantly reduced. It may be necessary to look into new manufacturing concepts as well as materials. For example, it may be cost-effective to adapt the auto industry's approach of a common basic airframe within which many models are possible. The use of low cost materials, such as plastics and styrofoam products, could result in low cost, and therefore expendable vehicles which would alleviate the shipboard recovery problems. The adaptation of plastic racing car engine technology is another area of opportunity.

Avionics- Avionics is one area where recent technological advancements hold the potential for significant improvements in size reduction, LCC and operational flexibility and effectiveness. VHSIC technology, BIT, AI and multi-sensor data fusion and processing techniques, coupled with high speed fiber optics data bus architecture, the availability of JTIDS and GPS all could significantly contribute to increased RPV applications. Again, the cost of applying these technologies must be assessed in the context of the total system, and a means for cost-effective technology transition from aircraft systems to RPV systems must be developed.

Safety- Shipboard safety considerations pose a unique need for technology advancements. The use of exotic fuels must be avoided. Otherwise, special storage and handling facilities must be integrated into the ship and be included in the cost of the RPV system. Recovery of the RPV onboard ship must provide for positive identification of the RPV in hostile or EMCON conditions, positive terminal guidance, and safe abort/go-around capabilities.

#### SUMMARY

Navy technology requirements for unmanned airborne vehicles are most unique in their shipboard launch, recovery and maintainability. Certainly the Navy has similar needs to those of the Army and Air Force for advanced sensors, electronic warfare, communications, avionics and airframe technologies, and it is not intended that these be overlooked. The Navy's current and future RPV programs will demand cost and mission effective systems with a high degree of survivability. It is, however, the Navy's intention to emphasize cost-effectiveness from the shipboard operation end of the problem, starting with storage,

launch, remote piloting and data acquisition, through the tradeoff of expendability vs recovery, and shipboard maintenance and ILS concepts.

It must be recognized that the overall effectiveness of an RPV system is not likely to be as high as that of a manned aircraft system because, among other factors, the absence of a crew precludes rapid response/counterresponse to changing or unpredictable circumstances. RPV systems, therefore, must demonstrate an extremely high degree of effectiveness in some missions or particular mission tasks and at much lower cost in comparison to manned systems. Only upon achieving these criteria will RPVs become an integral part of the Navy's airborne assets with a definable role and doctrine to govern their routine deployment from a wide range of ship classes.

This paper has shown that the Navy's reliance on manned aircraft is not entirely based on institutional bias. RPVs have much to offer. However, until the technology can provide RPVs fully integrated with shipboard operations, Navy applications will remain the same.

TABLE 1  
POTENTIAL NAVY RPV MISSION APPLICATIONS

<u>ATTACK</u>	<u>ANTI SURFACE WARFARE</u>
<ul style="list-style-type: none"> <li>○ INTERDICTION</li> <li>○ CLOSE AIR SUPPORT</li> <li>○ DEFENSE SUPPRESSION/DILUTION</li> <li>○ STRATEGIC SUPPORT</li> <li>○ ANTI-SHIP</li> <li>○ ELECTRONIC COUNTERMEASURES</li> <li>○ FORWARD AIR CONTROLLER</li> </ul>	<ul style="list-style-type: none"> <li>○ OTH DETECTION CLASSIFICATION AND TARGETING</li> <li>○ AREA (OCEAN) SURVEILLANCE</li> <li>○ DECOY</li> <li>○ PSA</li> <li>○ MINING</li> </ul>
<u>RECONNAISSANCE</u>	<u>ANTI-AIR WARFARE</u>
<ul style="list-style-type: none"> <li>○ AREA SEARCH/COVER</li> <li>○ ROUTE/STRIP SEARCH</li> <li>○ PINPOINT COVER</li> <li>○ TACTICAL WEATHER</li> <li>○ STRIKE CONTROL AND RECONNAISSANCE</li> <li>○ COMBAT RECONNAISSANCE</li> </ul>	<ul style="list-style-type: none"> <li>○ STRIKE ESCORT</li> <li>○ FIGHTER SWEEPS</li> <li>○ TARCAP</li> <li>○ BARCAP</li> <li>○ MISCAP</li> <li>○ DECK LAUNCH INTERCEPTOR</li> <li>○ BEACH HEAD AIR DEFENSE</li> </ul>
<u>COMMAND, CONTROL, COMMUNICATION</u>	<u>SURVEILLANCE</u>
<ul style="list-style-type: none"> <li>○ AIRBORNE EARLY WARNING (AEW)</li> <li>○ AIRBORNE RELAY</li> <li>○ BATTLEFIELD AIRBORNE COMMAND POST</li> </ul>	<ul style="list-style-type: none"> <li>○ OCEAN SURVEILLANCE</li> <li>○ ELECTROMAGNETIC SUPPORT MEASURES (ESM)</li> <li>○ COMINT</li> <li>○ CRISIS MANAGEMENT</li> <li>○ AIRBORNE SYSTEM SURVEILLANCE AND TRACK</li> <li>○ IMPLANTED SENSOR READOUT</li> </ul>
<u>SUPPORT</u>	<u>ANTI-SUBMARINE WARFARE</u>
<ul style="list-style-type: none"> <li>○ CARRIER ON-BOARD DELIVERY (COD)</li> <li>○ VERTICAL AIRLIFT</li> <li>○ FIXED WING AIRLIFT</li> <li>○ EMERGENCY RESUPPLY</li> <li>○ AERO MEDICAL EVACUATION</li> <li>○ RESCUE</li> <li>○ INFILIGHT REFUEL</li> </ul>	<ul style="list-style-type: none"> <li>○ CONTACT INVESTIGATION</li> <li>○ PERSISTENT SEARCH AND ATTACK</li> <li>○ SUBMARINE TRANSIT IDENTIFICATION ZONE (STIZ) ESTABLISHMENT AND MONITORING</li> <li>○ DEFENSIVE MISSIONS</li> <li>○ MINING OPERATIONS</li> <li>○ SURFACE SURVEILLANCE AND ATTACK (SSA)</li> </ul>

		RPV CLASS		
		SHORT-RANGE	MID-RANGE	LONG-RANGE
SHIP CLASS	SMALL COMBATANTS	X		
	CAPITAL SHIPS	X	X	
	LARGE CARRIERS		X	X

TABLE 2: A POSSIBLE APPROACH TO SHIPBOARD BASING OF NAVY RPV's

DESIGN/DEVELOPMENT TRADEOFFS TO ACHIEVE A MISSION EFFECTIVE  
SOW GUIDANCE SYSTEM

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SUMMARY

The increasing role of Precision Guided Standoff Weapons (PGSOW) places unique requirements on the navigation systems conventionally provided to this class of vehicles. Furthermore, the standoff mission requires innovative system concepts altogether different from the conventional system configurations that have become almost a "standard" approach to fighter and bomber navigation systems. This need for innovation is even more necessary if the SOW navigation system is to be affordable and mission effective in the context of these small relatively low cost vehicles.

The paper develops and reviews the navigation requirements, identifies and performs a trade-off of the alternatives as a function of guidance effectiveness.

From the spectrum of navigation system alternatives the author develops the specifics of a multiple sensor self-contained system showing the benefits to be derived by an optimum integration of low cost sensors.

The paper concludes with a description of the key elements of this integrated multiple sensor navigation system.

1. INTRODUCTION

The successful delivery of a precision guided standoff weapon (PGSOW) depends, in general, on three key elements, namely, knowledge of launch position parameters, knowledge of target position and accuracy of navigation between these two positions. The current state of the art in electronic navigation systems can likely provide all of the precision which is needed for successful SOW delivery. The freedom of choice among achievable navigation systems however is usually inhibited by practical considerations, chief among which may be cost and size. This is especially true for the navigation of the weapon from launch to target because the high quantity tactical SOW cannot support a high cost and size burden for this function.

This paper addresses the choices for solving the launch to target window phase of the navigation problem for a PGSOW.

2. NAVIGATION REQUIREMENTS FOR PGSOW DELIVERY

The basic PGSOW mission scenario is shown in Figure 1 which depicts the flight legs from aircraft takeoff to weapon launch point, then the weapon flight to target area window and finally to the impact point.

Among the essential parameters which control the success of the weapon delivery are the accuracy with which the launch position is known, the error in knowledge of the target position, and the accuracy of the navigation of the weapon itself. The first and second of these parameters depend on the accuracy of the aircraft's navigation and/or fix-taking systems and the accuracy of the intelligence concerning the target. These subjects are beyond the scope of this paper and for our purpose it will be convenient to assume that the errors of aircraft and target positions are not significant. This assumption is implicitly justified by reported delivery accuracy goals of several meters using sophisticated inertial navigation for PGSOW weapons. (Aviation Week, Oct. 17, 1983, p. 54)

The third parameter is addressed in this paper in terms of choices for providing accurate navigation to the low cost short (20 mile) range RPV or powered glide bomb which cannot sustain the size, weight or cost penalties of a high precision navigation system.

In order to ascertain the accuracy requirement for the PGSOW it is necessary to estimate the capability of the terminal seeker which in the final analysis, determines the size of the target window or search area. The burden placed on the terminal seeker increases with the size of the required search area imposed by uncertainty in relative weapon and target positions. In the case of an IR seeker, for example, the window may be relatively narrow for a given angular field of view because of the limited range due to atmospheric attenuation. An active millimeter

wave radar can provide longer range operation in bad weather (several kilometers) however the limited available aperture for the antenna and the consequent limited angular resolution creates problems with false alarms and poor detection probability as the search area is increased.

Therefore the PGSOW must be navigated to the target zone with sufficient cross track accuracy to insure that the target will lie within the swath described by its forward motion and the scanning limits of the terminal seeker. Based on the foregoing types of considerations it can be concluded that a practical search area width may be taken as being about 1 km. The corresponding navigation cross track error budget should be limited to one-half of this, or 0.5 km.

The along track error is dependent on the allowable false alarm and desired detection probability criteria. Obviously, as the along track extent of the search decreases these probabilities become more favorable. For the moment, however, it would be convenient to specify the along track navigation error as being the same as the cross-track, namely 0.5 km. It will be seen in what follows that the achievable along track accuracy as provided by a Doppler radar is significantly better.

To achieve the above performance objectives, the navigation system for the PGSOW must have a heading error over the 20 miles of better than 1 degree and an along track error of less than 1.5 percent of distance travelled.

### 3. NAVIGATION SYSTEM CHOICES

In addition to the criteria of accuracy, size and cost, the post launch navigation system must meet a number of other requirements. It should have an accuracy characteristic which is independent of the time of flight in order to accommodate slow flying weapons. Also, because tactical weapons are characteristically stockpiled for long periods without attention, it should have a long shelf life without need for periodic maintenance.

Principal among the available radio aids to navigation are the external radio techniques such as LORAN, OMEGA and GPS. However, if the navigation system depends on external radio aids, it may be denied their use by local electronic countermeasures. Therefore it would be very desirable to use a system which is autonomous and highly resistant to ECM. Furthermore, a fast flying missile needs to know velocity, attitude and heading for navigation and pilotage which these external radio aids do not provide directly since they are essentially fix-taking systems which provide position information at a specific instant in time.

Alternatives to these fix takers are dead reckoning systems which integrate the outputs of heading and velocity sensors to continuously update position, while using the output data for vehicle control and pilotage.

The most basic autonomous self-contained ECM resistant dead reckoning navigation system would consist of a magnetic compass and an air speed sensor, plus computer to compute the vector distance travelled by the vehicle. Such a system, although low in cost, would be grossly inadequate for the PGSOW application because of the large compass errors and the very large errors in ground speed introduced by unpredictable air mass movement (wind) and therefore will not be considered further in this paper.

At the upper end of the spectrum of sophistication is the high quality inertial navigation system which can readily meet the performance requirements but which is likely to be incompatible with the cost goals for tactical PGSOW's.

However the state of the art of velocity, heading and attitude sensors now provides options which, by judicious selection and combination, can provide a low cost dead reckoning system to satisfy the selected PGSOW mission. The key sensors are shown in Figure 2. These are the heading reference, attitude reference, Doppler velocity sensor, computer and angular rate sensors.

Five navigator configurations based on this elementary model will be considered, all using a Doppler radar velocity sensor to provide true ground velocity.

- o Pendulous flux valve, vertical gyro and three rate sensors
- o Same as preceding but with the addition of a directional gyro
- o Three-axis strapdown magnetometer, vertical gyro and three rate sensors
- o Three-axis strapdown magnetometer, two accelerometers and three rate sensors
- o Three-axis strapdown magnetometer, two-axis gyro-accelerometer sensor and a single rate sensor.

#### 3.1.1 DOPPLER RADAR, PENDULOUS FLUX VALVE, VERTICAL GYRO AND THREE RATE SENSORS.

A block diagram of this configuration is shown in Figure 3.

In each of the configurations which are to be discussed, a Doppler radar velocity sensor is the sensor of choice. A Doppler velocity sensor was selected because it

operates in all weather, provides very accurate velocity, is small in size, is highly ECM resistant and now available at low cost. It also provides radar altitude. (The specifics of the radar will be described in a later section.)

A low-cost vertical gyro is capable of providing the required pitch and roll accuracy of 0.5 degrees ( $1\sigma$ ) but only in the absence of acceleration. The typical pitch and roll accuracy of a low-cost vertical gyro during low-level maneuvers is 3 degrees ( $1\sigma$ ).

The rate sensors shown in this configuration are generally required for autopilot operation and can be utilized to decouple the "g" sensitive elements of the heading system.

The pendulous flux valve is probably the simplest method of obtaining heading. However, the accuracy of the heading output is also the poorest. The flux valve measures the horizontal components of the earth's magnetic field along two orthogonal axes and provides signals proportional to the sine and cosine of heading. The flux valve is capable of making these measurements accurately in the absence of aircraft maneuvers and accelerations. However, when the aircraft is maneuvering, the flux valve tilts from its horizontal position, and the measurements of the horizontal components include a vertical component of the earth's magnetic field. This results in errors in the heading measurement which can be substantial. Figure 4 shows the heading error versus dip angle<sup>1</sup> of the earth's magnetic field for various attitude errors of the flux valve; e.g., a tilt of 2 degrees and a dip angle of 70 degrees would result in a heading error of more than 5 degrees that varies sinusoidally with heading. The error can be minimized by not using flux valve data in short duration turns and substituting remembered heading plus integrated heading rate. However, during sustained turns rate sensor bias errors would be prohibitive.

The impact of heading errors on the accuracy of navigation is shown in Figure 5. This figure supports the need for a 1.0 degree heading accuracy to achieve the 500 meter navigation window at 20 miles (32 km). Therefore it follows from Figure 4 that the attitude reference must be well within 0.5 degrees of vertical to achieve this with some allowance for the additional intrinsic error of the heading reference.

### 3.1.2 DOPPLER RADAR, PENDULOUS FLUX VALVE, DIRECTIONAL GYRO, VERTICAL GYRO AND THREE RATE SENSORS.

This configuration is shown in Figure 6 and is somewhat similar to that of Figure 3, the difference being the addition of the Directional Gyro (DG). This configuration is one which is typically used in conventional magnetic heading references e.g., the U.S. Army AN/ASN-43. A directional gyro alone provides good short term heading data and in the absence of gyro drift would provide accurate heading information after it had been initially aligned to magnetic north. However, gyro drift results in gradual degradation in the accuracy of the heading data, therefore, the heading output of the directional gyro is slaved to the flux valve output. The effective time constant of the flux valve - DG loop is made very long to filter the flux valve noise and to provide a slow response of the DG to flux valve errors. Additional improvement can be obtained by cutting out the flux valve data during maneuvers when the errors become large. The result is a heading reference which has a good short term response and is effectively limited by the accuracy of the flux valve during stable flight.

This heading reference is generally capable of providing 1 degree ( $1\sigma$ ) heading accuracy only in the absence of extended severe maneuvers. In addition, the system suffers from the additional cost and weight penalty of the DG.

Attitude and angular rate considerations for this configuration are identical to those for the Figure 3 configuration.

### 3.1.3 DOPPLER RADAR, THREE-AXIS STRAPDOWN MAGNETOMETER, VERTICAL GYRO AND THREE RATE SENSORS.

This configuration is shown in Figure 7. The attitude and rate sensors are identical to those discussed in 3.1.1 and 3.1.2. However, in this case the technique for measuring heading is different in that it uses a magnetometer.

The magnetometer measures the three components of the earth's magnetic field along the airframe longitudinal, drift and vertical axes. The pitch and roll data provided by the vertical gyro are then used to perform coordinate transformations of the magnetic field data into a horizontal plane. Heading is then computed from the two horizontal components of the earth's magnetic field.

The accuracy of the heading achievable using this configuration is limited primarily by the accuracy of the pitch and roll data of the vertical gyro. The discussion in 3.1.1 indicated that a deviation from vertical of the flux valve of as little as 2 degrees resulted in a heading error of greater than 5 degrees when the

<sup>1</sup>Dip angle is defined as  $\tan^{-1} B_v/B_h$ , where  $B_v$  and  $B_h$  are the vertical and horizontal components of the earth's magnetic field.

dip angle of the earth's magnetic field is 70 degrees. The same results are obtained when heading is obtained by transforming the outputs of a three-axis magnetometer using pitch and roll data that are in error.

A low-cost vertical gyro can provide accurate pitch and roll data in the absence of vehicle acceleration resulting in accurate heading data. However, during aircraft maneuvers, the level sensors in the vertical gyro respond to the maneuver-induced accelerations and tend to drive the pitch and roll outputs away from the true values. This configuration has an advantage over that of Figure 3 since the stabilization loops which are part of the vertical gyro act as long time-constant filters to the level sensor errors during maneuvers and this reduces the resulting vertical errors. The performance of this configuration is expected to be better than that of Figure 3 because it avoids the large errors associated with the pendulous flux valve during maneuvers. However, as discussed in 3.1.1, the typical pitch and roll accuracy of a low-cost vertical gyro during low-level maneuvers is 3 degrees ( $1\sigma$ ), and this level of performance, although better than that of the pendulous flux valve, still would not result in heading of the desired accuracy using the configuration of Figure 7.

#### 3.1.4 DOPPLER RADAR THREE-AXIS STRAPDOWN MAGNETOMETER, TWO ACCELEROMETERS AND THREE RATE GYROS.

This configuration is shown in Figure 8. In this configuration, a different technique is used to obtain pitch, roll and heading.

The accelerometers provide acceleration data along the aircraft X and Y axes and the rate sensors provide angular rates about the aircraft X, Y and Z axes. These data plus the Doppler velocity data are combined to form a Doppler-Inertial (D-I) loop in software to provide accurate pitch and roll even in the presence of aircraft maneuvers. The pitch and roll data thus derived are then available for use by the autopilot and also to perform the coordinate transformations of the 3 axis strapdown magnetometer data through pitch and roll to obtain heading. The improved pitch and roll will result in improved heading accuracy.

Figure 9 is a single-axis block diagram of a simplified Doppler-Inertial (D-I) mechanization. The accelerometer measures both acceleration and the component of gravity due to tilt. Angular rate is measured by the rate gyro, compensated for earth and vehicle velocity, and used to compute the component of gravity that is subtracted from the accelerometer output. The compensated acceleration is integrated to form an estimate of "inertial" vehicle velocity that is compared with the corresponding component of Doppler velocity. The difference of these two velocities is fed to a Kalman filter that determines the error in tilt and also generates a "D-I" velocity. Alternatively a simplified filter using constant gains in the state vector matrix to reduce computational load could be used to obtain these data. Fixed gains are a reasonable compromise when the vehicle trajectory is not changing significantly during a mission. The corrected tilt angle is then used when computing the horizontal components of the earth's magnetic field.

In steady-state operation of the D-I loop only the bias errors of the accelerometers produce any pitch or roll errors, whereas Doppler and gyro bias errors result in zero pitch and roll errors. Doppler and gyro random noise do cause pitch and roll errors, and optimum performance is achieved by the careful selection of the loop gains. These gains affect the loop response speed and damping characteristics. A fast response reduces pitch and roll errors due to rate sensor noise but increases the errors caused by Doppler noise. Thus the choice of these parameters is a compromise to achieve a minimum total error. The computational requirements for this configuration are greater than those of the previous configurations, but can be met by currently available microprocessors such as the Intel 8086.

This configuration, while having the most sensitive components, provides very acceptable pitch, roll and heading data even under severe maneuver conditions.

#### 3.1.5 DOPPLER RADAR THREE-AXIS MAGNETOMETER, MULTISENSOR GYRO-ACCELEROMETER AND A RATE SENSOR.

This configuration is shown in Figure 10 and is similar to that of Figure 8. In this case, a unique gyro-accelerometer device, called a MULTISENSOR<sup>®</sup> which will be described later, outputs four signals: acceleration along the X and Y axes of the aircraft and angular rates about these same axes, thus incorporating into one unit the functions provided by four separate sensors in the Figure 8 configuration. Angular rate about the aircraft Z-axis is provided by a separate rate sensor.

Mechanization and operation of this configuration is similar to that of Figure 9 and simulation studies based on the configuration using a Kalman filter to perform the D-I combining have been performed (see paragraph 5). The results indicate that pitch and roll accuracy of 0.17 degree rms are achievable even during maneuvers. A significant factor in the improved performance of this configuration is the low random gyro drift rate of the MULTISENSOR<sup>®</sup> of 2 degrees/hour as compared to the 50 to 100 degrees/hour random drift rates of low-cost single-axis rate gyros. As mentioned earlier the effects of bias gyro drift are eliminated in this D-I mechanization. In

addition D-I mixing reduces overall cost by eliminating gyro calibration time and the hardware usually required to store the calibration constants.

### 3.1.6 SUMMARY.

A trade-off comparison chart for the above five systems is shown in Table I. The characteristics compared are performance, cost, size and weight. The Figure 3 system configuration is considered the baseline for comparison. The table shows that the MULTISENSOR™ rate sensor configuration shown in Figure 10 is superior to all the other configurations in all respects including performance, cost, size and weight. Thus, even for those applications in which the operating environment is rather benign so that acceptable performance is achievable with any of the configurations, this configuration may likely be the most desirable.

Use of the MULTISENSOR™ in place of a vertical gyro and two of the three rate gyros would change the partitioning of several functions. Specifically, a single multiplexed A/D converter could be used for all analog inputs including air data inputs should they be available and required by the autopilot. Computations required for the inertial elements and magnetometer could be performed in an upgraded microprocessor in the Doppler Velocity Sensor (DVS) or in a central computer unit. To eliminate duplicative hardware regulated voltages and timing signals and various clocks could be provided by the DVS. The inertial sensors could be packaged on the DVS to minimize volume and supports while assuring accurate alignment to the DVS antenna.

In summary, the advantages of the Figure 10 configuration are:

- o Provides accurate velocity, pitch, roll, heading and attitude even under severe maneuver conditions
- o Provides short term inertial velocity during any period where Doppler may be in memory, such as in steep dives
- o Provides superior angular rate data about two axes
- o Provides lowest overall system cost achieved by combining functions of several sensors
- o Provides smallest volume and weight.

## 4. DISCUSSION OF NAVIGATION SYSTEM ELEMENTS

Neither a Doppler Radar Velocity Sensor nor an integrated inertial sensor are normally perceived as low cost devices. However, the state of the art in both sensor areas has moved forward with the application of new technology and advanced concepts so that cost and size have come down remarkably with performance being maintained or improved over older technologies. We will first describe a modern Doppler velocity sensor, then a unique gyro-accelerometer all-in-one device, and finally the magnetometer, all of which are eminently suited to the PGSOW mission.

Finally, flight results with a magnetometer will be described.

### 4.1 DOPPLER RADAR VELOCITY SENSOR (DRVS).

Figure 11 is a photograph of a Ku-band DRVS which measures 12" x 7" x 2 1/2" in a package which contains the transmitter, receiver, beam velocity processor, power supply and antenna for a total weight of 7 pounds. It provides three components of velocity in deck plane coordinates, namely along heading ( $V_x$ ), cross heading ( $V_y$ ) and velocity normal to the deck plane ( $V_z$ ) as well as radar altitude measured along each of the four beams. The low size and cost achieved in this unit is made possible by the use of microwave integrated circuits, VLSI in the signal processor and I/O and a printed microstrip antenna.

The elements of the DRVS are shown in Figure 12. The antenna is directly mounted to the case making it an integral part of the LRU. Its radome forms the bottom surface of the LRU. The transmitter is a Gunn diode oscillator consisting of two active components (a Gunn diode and a varactor diode) mounted in a tuned microwave cavity. The Gunn diode oscillates at a nominal frequency of 13.325 GHz. The varactor diode is energized with a 30 kHz sine wave modulating signal. The resulting frequency modulated-continuous wave (FM-CW) signal is radiated at a level of about 50 milliwatts from the printed antenna.

Kearfott's microstrip antenna is a major factor in achieving a very thin, low-cost and lightweight Doppler radar. The antenna consists of a grid of metal "patches" etched onto a dielectric substrate that is bonded in turn to a metal back plate. The grid and back plate form a two-dimensional array of radiating elements designed to result in a narrow and well-defined beam of radiation with very low sidelobes. Beams in several different directions are obtained by switching the excitation to different feedports on this array of radiators.

The cost of this DRVS is approximately an order of magnitude lower than the latest current operational DRVS. Its layout, architecture and technology permit dramatic

savings in production costs at no sacrifice in performance, which has been repeatably demonstrated to have an accuracy of 0.3 percent of distance travelled. This is five times as good as was required by the discussion in Section 2. This permits reduction of the target search area with direct benefit to probability of detection and false alarm rate. The DRVS will operate over an altitude range from zero to 10,000 feet and a speed range from essentially zero to 600 knots.

The Doppler radar has a built in altimeter capability which provides altitude along each of the four antenna beams to an accuracy of  $\pm 5$  percent. Altimeter data can be useful in controlling the PGSOW during its flight to maintain clearance over the terrain.

#### Resistance to countermeasures.

The Doppler radar navigator is highly ECM resistant by virtue of its antenna directionality, coherence of its signal and ability to operate in a memory mode.

The Doppler antenna radiates four narrow beams symmetrically disposed in a downward direction. Thus, in flight, any one of the beams could point at a jamming source which was being overflowed for only a very brief period during which the Doppler might be driven into memory and would use the last measured value of velocity until the jammer was out of the Doppler antenna beam.

For directions outside of its main beams the Doppler antenna has a high signal rejection relative to the main beam. This directionality, coupled with the narrow bandwidth of the Doppler processor will foil all but the most powerful jammers in very close proximity to the PGSOW flight path.

The other elements of the system, i.e., the magnetometer, inertial sensors and computer are not susceptible to ECM.

#### Magnetometer.

The preferred configuration uses a triad of strapdown magnetometers in place of a conventional pendulously-suspended flux valve. As noted earlier the flux valve is self-stabilized to the local vertical and is very sensitive to vehicle accelerations. Three strapdown magnetometers measure each of the three orthogonal components of earth's magnetic field, and thus the two horizontal components (from which vehicle heading is computed) can be obtained by transforming through pitch and roll. The computed magnetic heading is thus independent of vehicle maneuvers to the extent that the externally provided pitch and roll are maneuver-insensitive.

Lightweight strapdown magnetometers are available from several sources including, for example, Sperry and Develco, Inc. Most units have DC outputs that can be converted into digital form. External effects that corrupt flux valve accuracy will also affect magnetometer accuracy. Soft and hard-iron effects, or magnetic deviation and uncertainties in magnetic variation have approximately the same effect and must be properly compensated if the overall system is to meet its navigation accuracy.

#### 4.4 MULTISENSOR GYRO-ACCELEROMETER DESCRIPTION.

Figure 13 shows the Kearfott MULTISENSOR<sup>®</sup> gyro-accelerometer. It is a unique multifunction inertial instrument which independently senses two axes of angular rate and two axes of linear acceleration. The instrument utilizes piezo-electric crystal flexure elements to measure the components of vehicle angular velocity and linear acceleration lying in a plane perpendicular to the instrument spin axis. A case referenced signal generator is used to relate the components measured in a rotating coordinate frame to instrument case fixed coordinate reference axes (x, y). Figure 14 is a functional representation of the angular rate and linear acceleration sensing arrangement. Physically, it is contained in a cylindrical case about 0.75 inch diameter by 1 $\frac{1}{4}$  inches long.

#### Angular rate.

The angular velocity sensor operation is based on the gyroscopic behavior of an elastically restrained body which is rotating at a high rate. Two piezo-electric crystal cantilever beams are arranged in a dipole configuration as shown in Figure 14. The piezo-electric beams act both as inertial members and the restoring spring. When an angular rate is applied in a plane orthogonal to the spin axis, the angular momentum of the spinning beams generates a suppressed-carrier modulated signal. The carrier is the spin frequency, and the amplitude of the signal is proportional to the magnitude of the input rate.

Figure 14 shows that an input angular rate causes one beam to deflect up and the other to deflect down. The crystal outputs are therefore opposite in polarity and are connected such that the signals are additive. An acceleration along the spin axis will cause the two beams to deflect in the same direction. However, in this case the net signal output is zero since the two crystal signals have the same polarity. Thus, the dipole arrangement provides common mode rejection, and the output signal is a function only of the angular rate input. The dipole arrangement also results in output signal enhancement.

### Linear acceleration.

The linear acceleration sensor assembly consists of the same piezo-electric cantilever flexure beam elements as in the rate gyro sensor assembly. However, a dipole configuration is used in which the sensitive beams are turned 90 degrees with respect to the gyro beams. In this orientation, the cantilevers will react to linear acceleration in the plane perpendicular to the spin axis. The spinning pickoff generates a suppressed carrier modulated signal (at spin frequency) with amplitude proportional to the input acceleration magnitude.

The Kearfott MULTISENSOR™ gyro-accelerometer, with only one rotating part, yields measurements equivalent to the output of four conventional single channel instruments. As a result, overall complexity and cost are considerably reduced. The open loop nature of the instrument offers an exceptional dynamic range of  $10^6$ , with maximum rate capabilities in excess of 1000 degrees/s and acceleration capabilities in excess of  $\pm 500$  g's.

### 5. SIMULATION STUDIES

Simulation studies of the proposed system have been performed using a general navigation system covariance simulation program developed by Kearfott. The purpose of the simulation was to evaluate the performance of the Doppler-aided MULTISENSOR™ and a single-axis rate gyro as a source of vehicle pitch and roll.

The results of these preliminary studies indicate RMS errors of 0.1 degree in pitch and roll during level flight with no accelerations and maneuver dependent transient errors up to 0.17 degree during turns and accelerations. Thus, it is reasonable to predict 0.25 degree RMS performance for the D-I loop.

This level of pitch and roll accuracy will enable accurate transformation of strapdown magnetometer outputs into horizontal components and hence enable accurate computation of magnetic heading.

### Flight tests.

The flight test configuration consisted of a single axis rate gyro to provide azimuth rate, a Sperry 3-axis magnetometer, and Kearfott's MULTISENSOR™ and Doppler radar. The outputs of all sensors were fed into a multiplexed A/D Converter and then to an HP-1000 computer. An F-16 Inertial Navigation System (INS) was used as the precision pitch, roll and heading reference for comparison with the pitch, roll and heading developed by the D-I system and the magnetometer. The computer performed all strapdown and D-I computations, and magnetometer coordinate transformations, and also comparisons of the "D-I" generated pitch, roll and heading data with the F-16 data.

Flight tests were performed in a DC-3 at an altitude of roughly 4000 feet and a speed of approximately 120 knots. The flights included straight and level operation as well as maneuvers with roll excursions up to 60 degrees and pitch angles of more than ten degrees. Figure 15 is a sample of the results obtained. The figure shows aircraft roll and pitch and the error in roll and pitch developed by the D-I system when compared with the F-16 INS data. The mean pitch and roll errors during these flight tests were very small - .036 degrees or less, and the rms errors ranged from .058 degree to .27 degree. Peak errors, even under the severe dynamic maneuvers, were no more than 1 degree, and rms heading errors were under 1 degree. The flight test verified the feasibility of configuring a guidance system with a heading accuracy of 1.0 degree ( $1\sigma$ ) while providing velocity, attitude, altitude and rate data.

### CONCLUSION

The choice of an autonomous self-contained navigation system is a low risk approach for solving the post launch PGSOW navigation problem based on the availability of sensors which have been proven by actual test and experience. Furthermore, each of the system elements, namely the Doppler radar, magnetometer, computer and MULTISENSOR™ are devices which lend themselves to high volume manufacture. The projected cost of such a combination of sensors is compatible with application to an expendable PGSOW of the "short range" class.

### References:

- (1) "USAF Studies New Standoff Weapons", Clarence A. Robinson, Jr., Aviation Week October 17, 1983.
- (2) The Multisensor Gyro-Accelerometer, A Low Cost Multifunction Strapdown Inertial Instrument for Tactical Applications", Thomas Pick, Institute of Navigation Symposium Record, January 17, 1984.

TABLE I. SUMMARY OF TRADE-OFF STUDIES FOR PITCH AND ROLL SENSOR

SYSTEM CONFIGURATION	MANEUVER CONDITIONS	ROLL AND PITCH	HEADING	ANGULAR RATES	COST	VOLUME* (in <sup>3</sup> )	WEIGHT* (lb)
1. PENDULOUS FLUX VALVE, VERTICAL GYRO AND 3 RATE SENSORS	MODERATE SEVERE	GOOD POOR	POOR	BIAS = 300°/h	BASELINE	164	6.5
2. PENDULOUS FLUX VALVE, DIRECTIONAL GYRO, VERTICAL GYRO AND 3 RATE SENSORS	MODERATE SEVERE	GOOD POOR	EXCELLENT GOOD	SAME AS (1)	SIGNIFICANTLY GREATER THAN BASELINE	314	11.0
3. 3-AXIS STRAPDOWN MAGNETOMETER, VERTICAL GYRO AND 3 RATE SENSORS	MODERATE SEVERE	GOOD POOR	GOOD	SAME AS (1)	APPROXIMATELY SAME AS BASELINE	106	4.0
4. 3-AXIS STRAPDOWN MAGNETOMETER, 2 ACCELEROMETERS, AND 3 RATE SENSORS	MODERATE SEVERE	GOOD GOOD	GOOD	SAME AS (1)	SIGNIFICANTLY LESS THAN BASELINE	14	1.1
5. 3-AXIS STRAPDOWN MAGNETOMETER, MULTISENSOR AND 1 RATE SENSOR	MODERATE SEVERE	EXCELLENT EXCELLENT	EXCELLENT	Z-AXIS - SAME AS (1) X, Y AXES - 10°/h	SIGNIFICANTLY LESS THAN BASELINE	13	1.0

\*EXCLUSIVE OF DOPPLER RADAR VELOCITY SENSOR, COMPUTER AND I/O (235 CU IN., 8 LBS.)

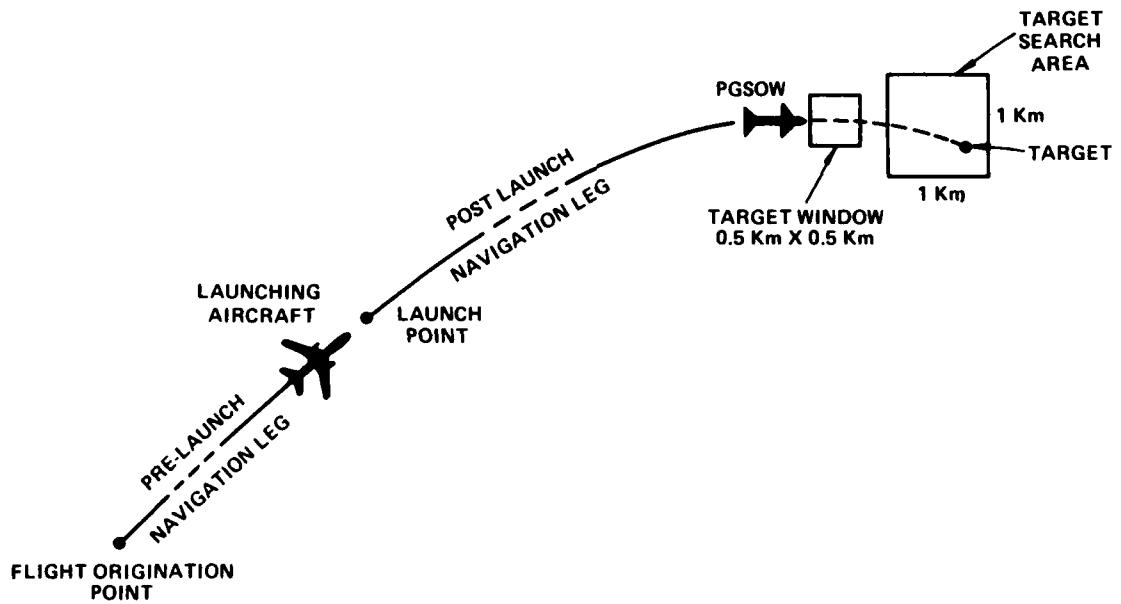


FIGURE 1. PGSOW MISSION SCENARIO

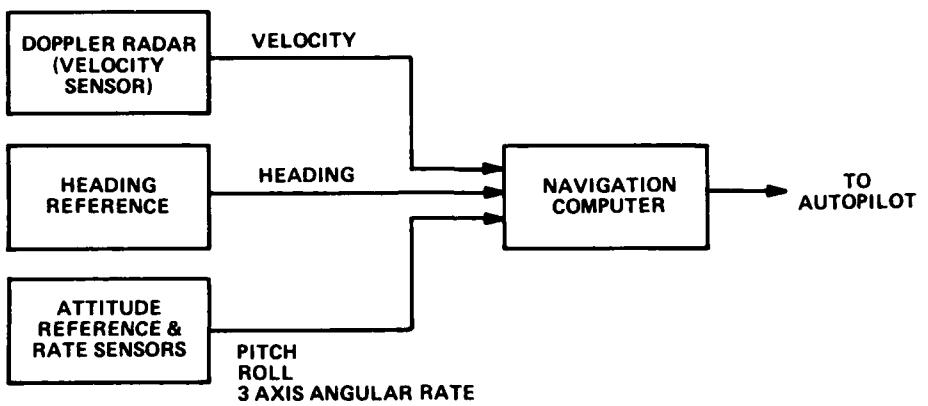


FIGURE 2. BASIC DEAD RECKONING SYSTEM

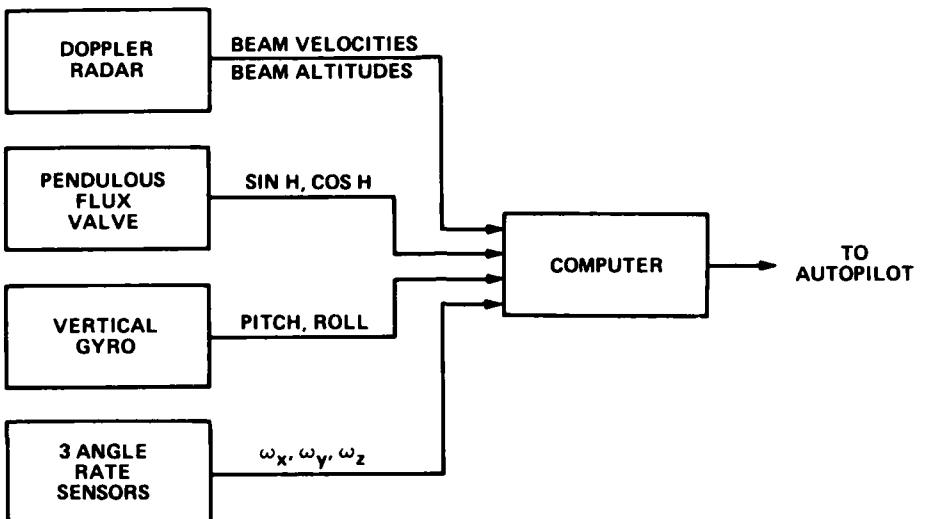


FIGURE 3. DOPPLER RADAR, PENDULOUS FLUX VALVE, VERTICAL GYRO AND THREE RATE SENSORS BLOCK DIAGRAM

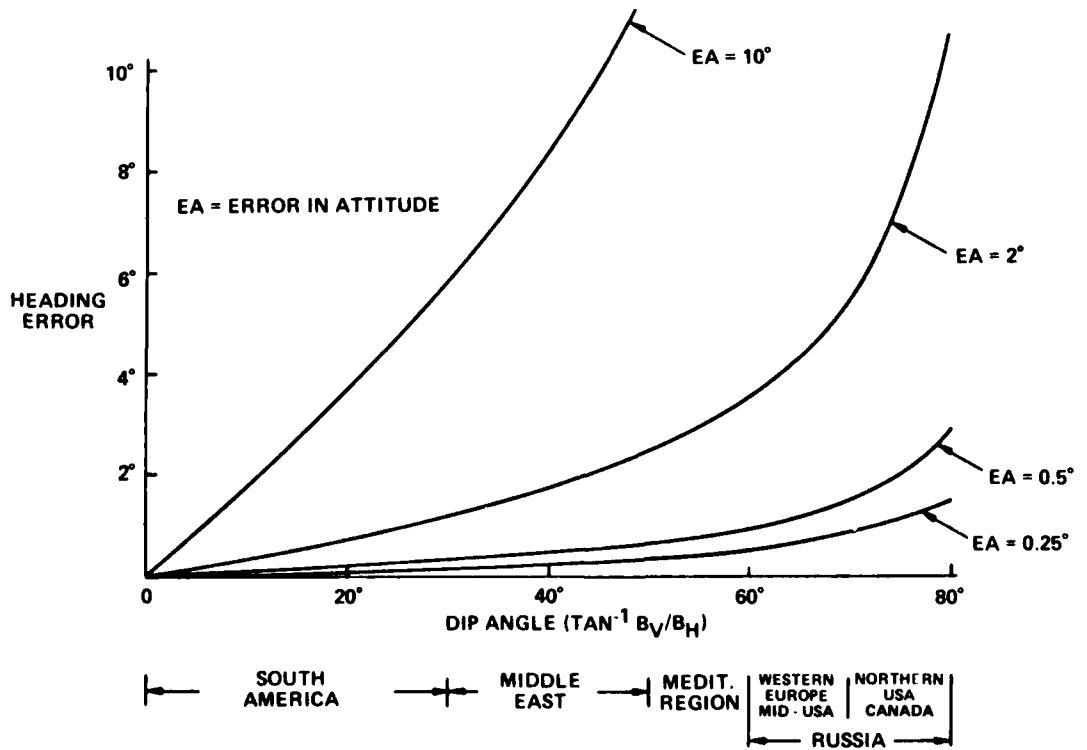


FIGURE 4. HEADING ERROR AS A FUNCTION OF DIP ANGLE AND VERTICAL ERROR

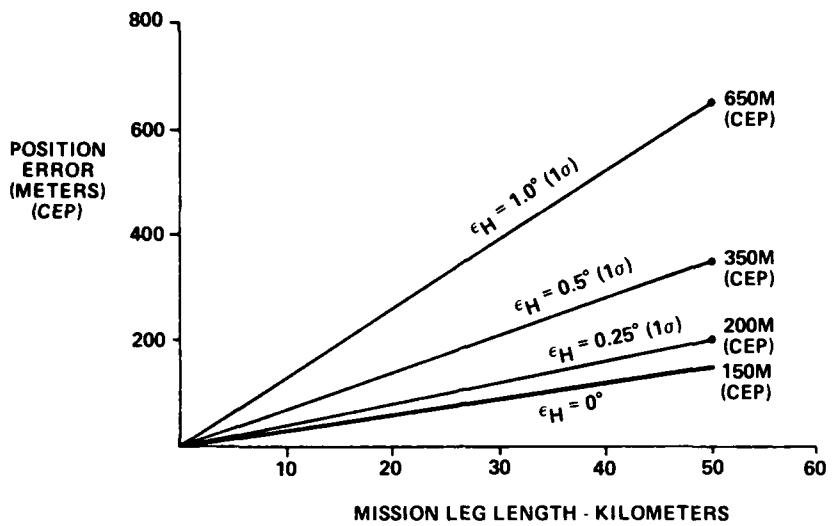


FIGURE 5. NAVIGATION SYSTEM TERMINAL ERROR

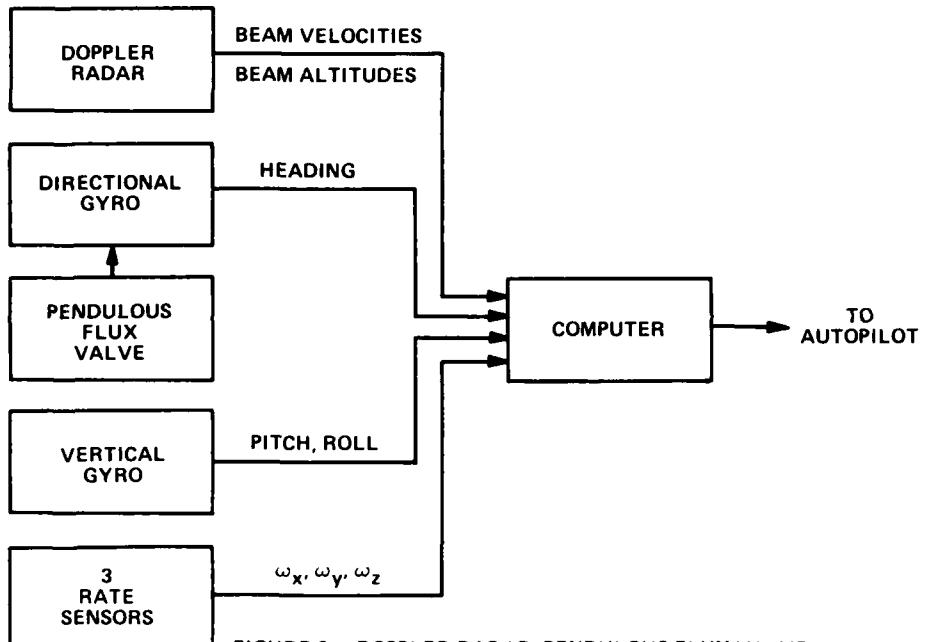


FIGURE 6. DOPPLER RADAR, PENDULOUS FLUX VALVE, DIRECTIONAL GYRO, VERTICAL GYRO AND THREE RATE SENSORS BLOCK DIAGRAM

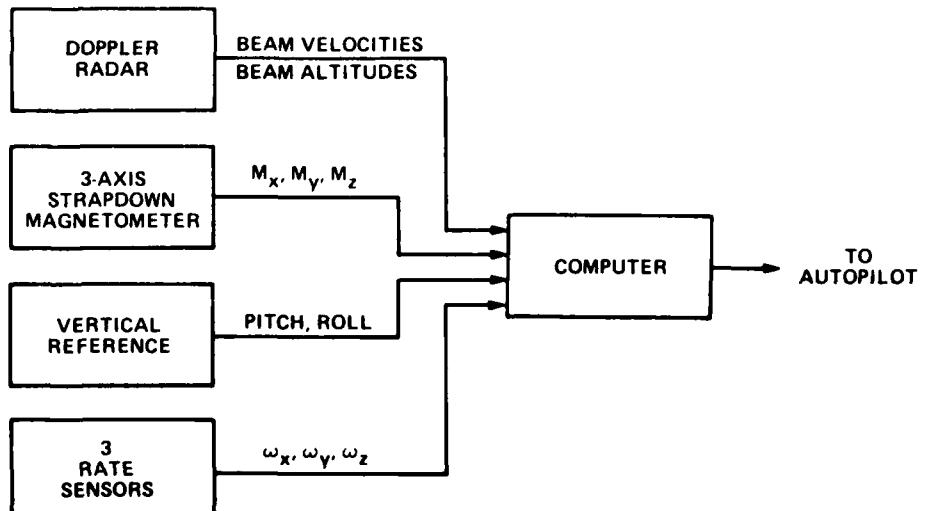


FIGURE 7. DOPPLER RADAR, THREE STRAPDOWN MAGNETOMETER, VERTICAL GYRO AND THREE RATE SENSORS BLOCK DIAGRAM

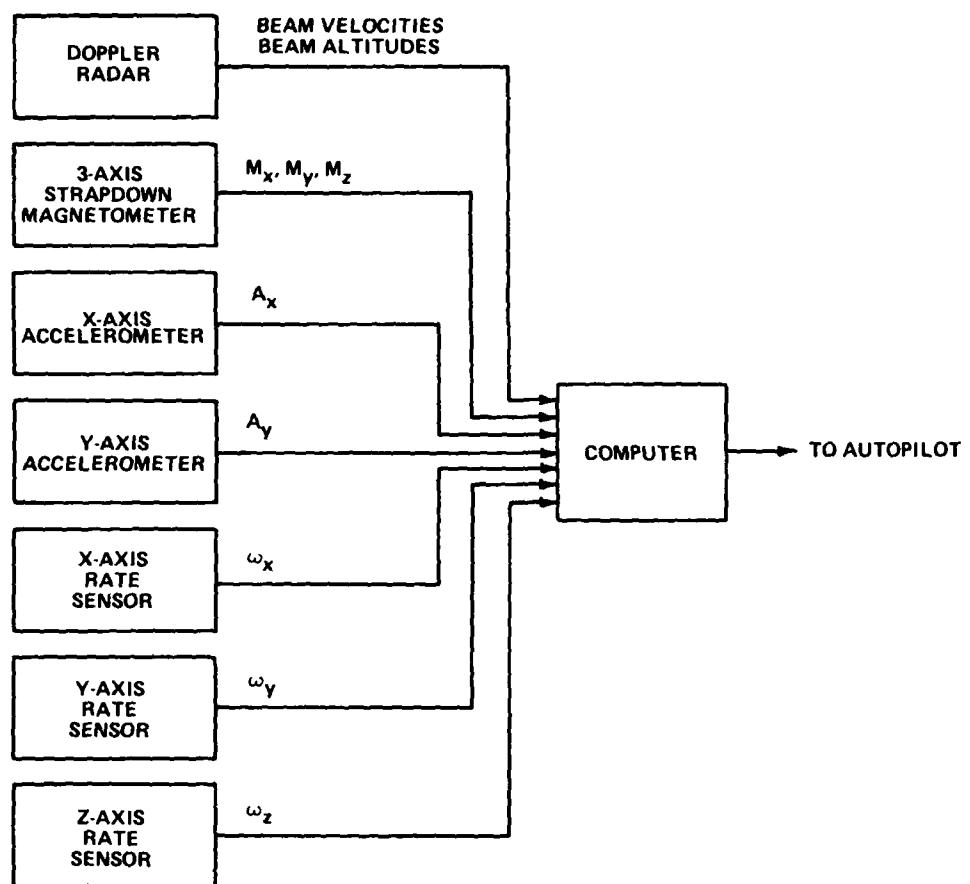


FIGURE 8. DOPPLER RADAR, THREE-AXIS STRAPDOWN MAGNETOMETER, TWO ACCELEROMETERS AND THREE RATE GYROS

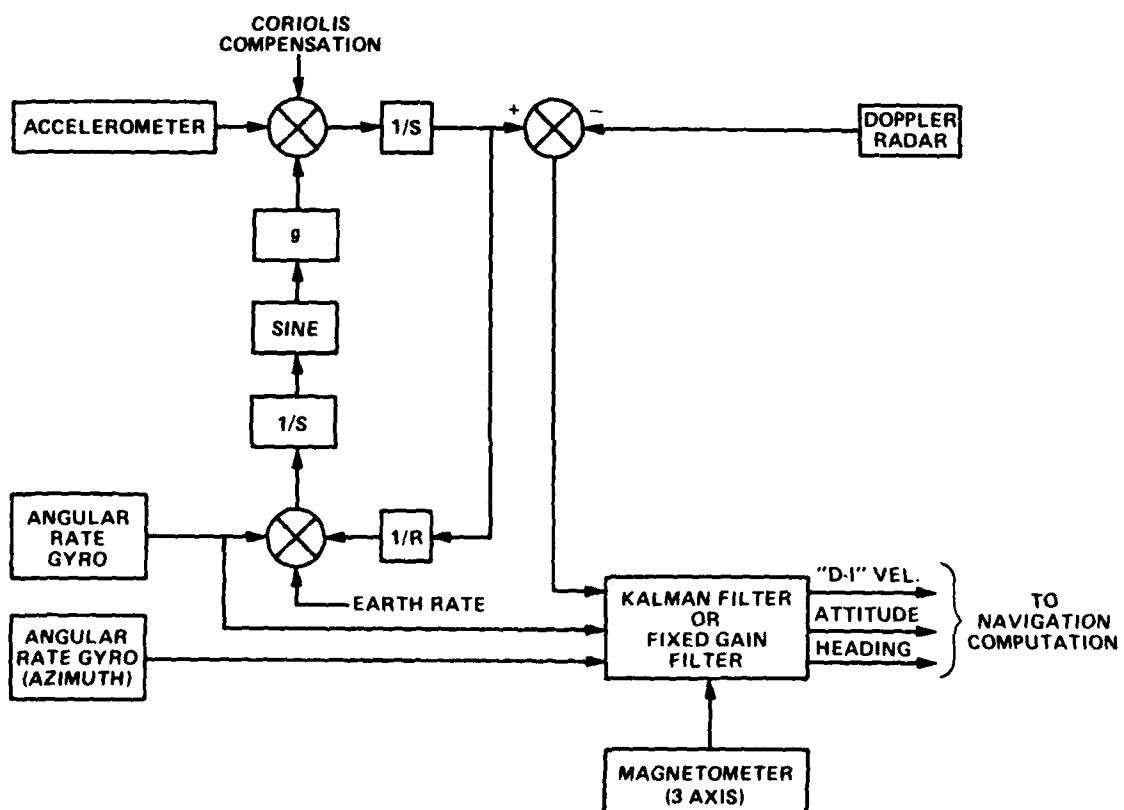


FIGURE 9. SIMPLIFIED DOPPLER - INERTIAL SYSTEM

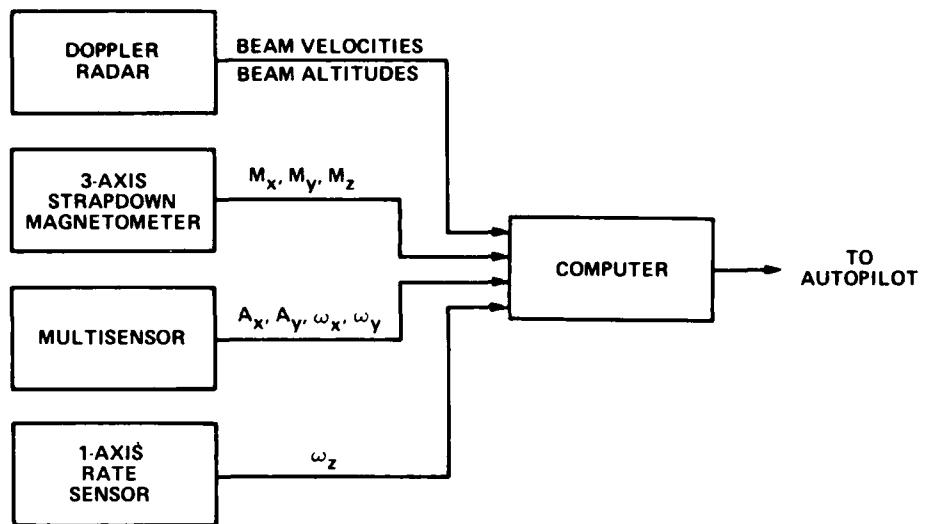


FIGURE 10. DOPPLER RADAR, THREE AXIS MAGNETOMETER,  
MULTISENSOR GYRO ACCELEROMETER AND  
A RATE SENSOR - BLOCK DIAGRAM

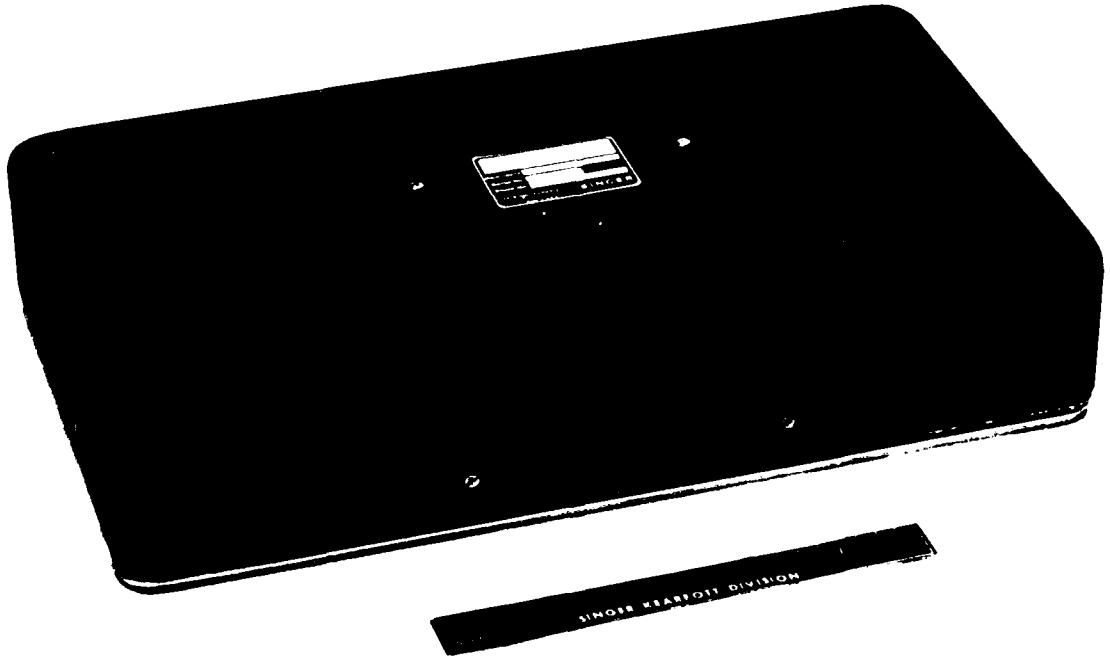


FIGURE 11. SKD-2900 DOPPLER RADAR  
VELOCITY SENSOR (DRVS)

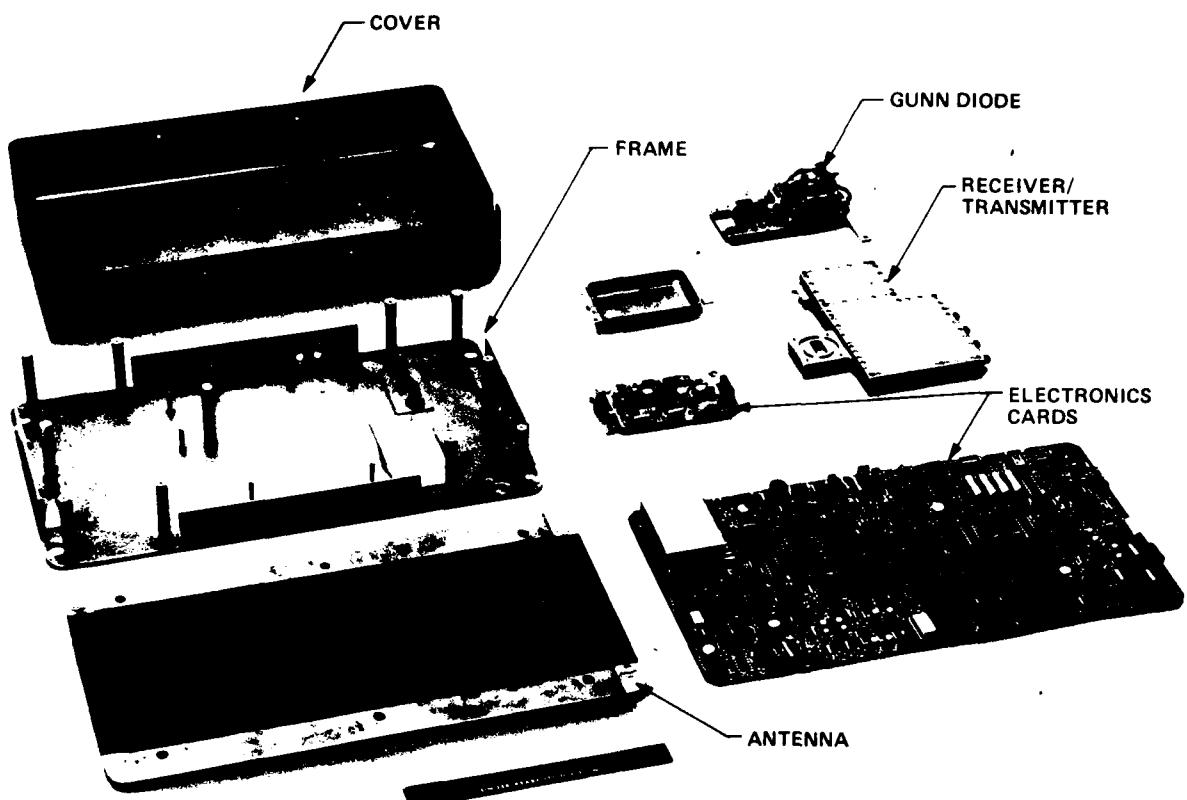


FIGURE 12. SKD-2900 DRVS (DISASSEMBLED)

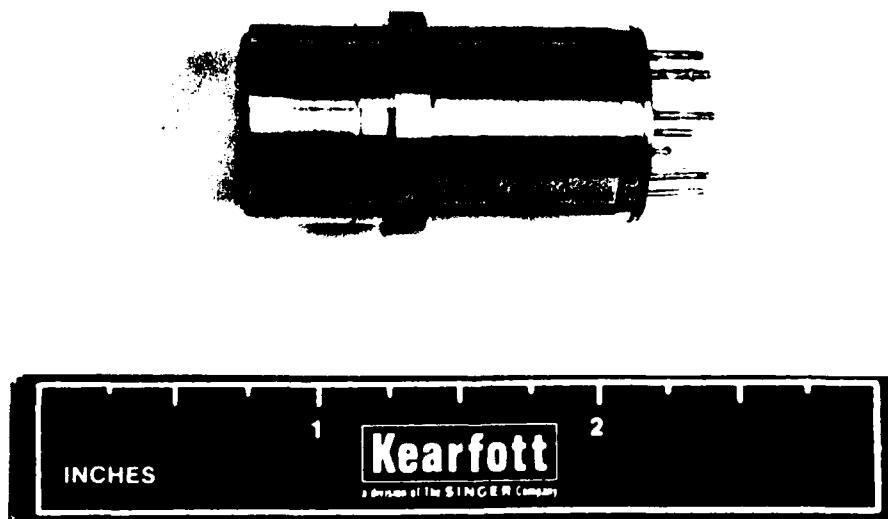


FIGURE 13. MULTISENSOR GYRO/ACCELEROMETER

BASIC PRINCIPLE

- DYNAMIC INPUTS TRANSLATED INTO FORCE
- RESULTING FORCE APPLIED TO PIEZO-ELECTRIC SENSOR
- SENSOR YIELDS PROPORTIONAL ELECTRIC SIGNAL
- ABILITY TO SENSE SPECIFIC DYNAMIC INPUT IS FUNCTION OF THE ORIENTATION OF BEAMS

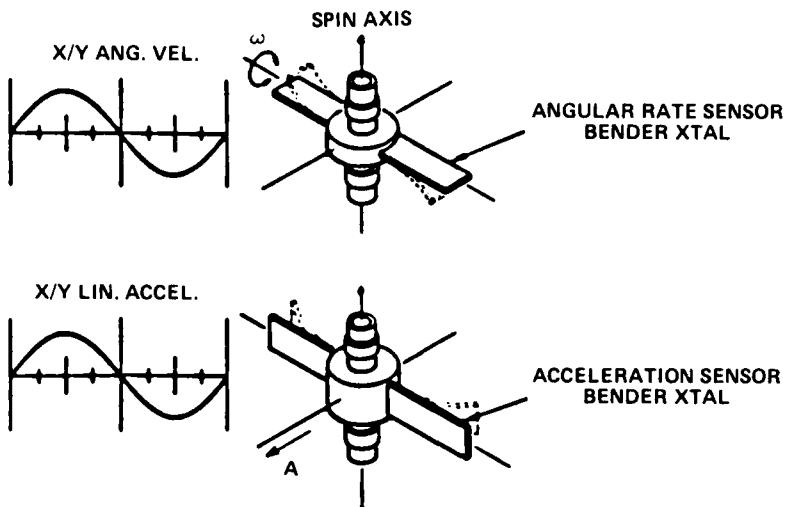


FIGURE 14. RATE AND ACCELERATION SENSING PRINCIPLES

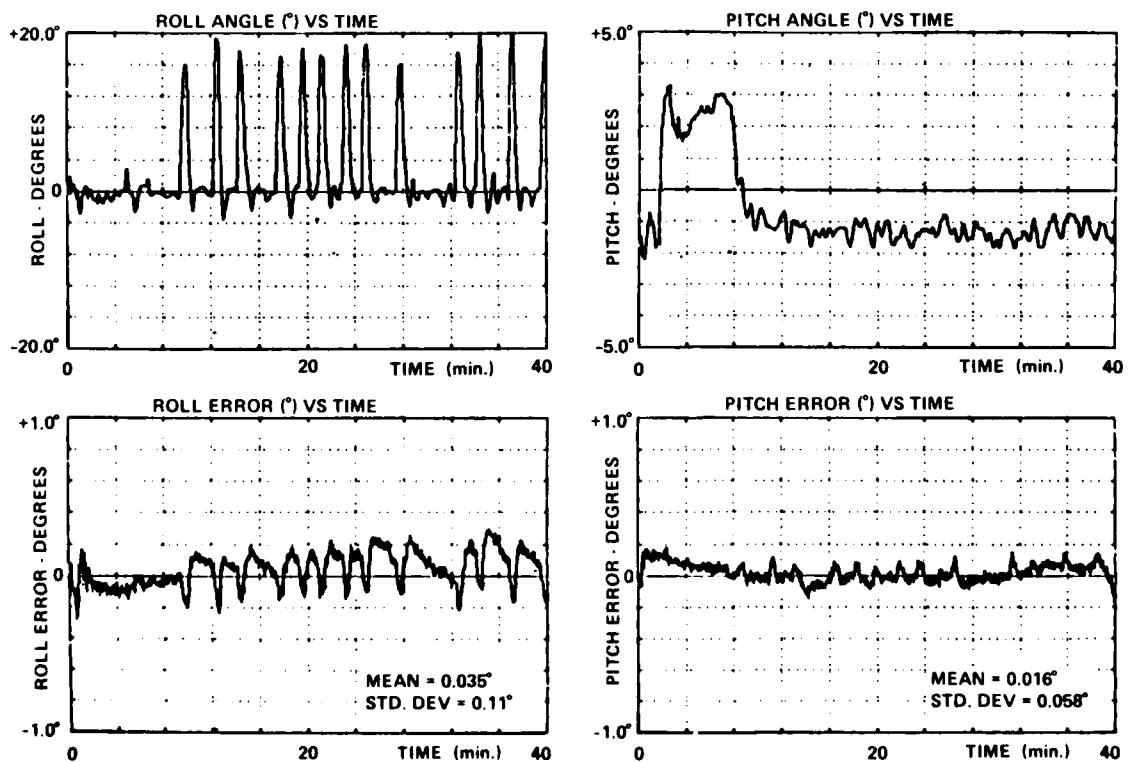


FIGURE 15. FLIGHT TEST RESULTS JANUARY 1984

## LETHAL AIR DEFENSE SUPPRESSION TECHNOLOGY BASE DEFINITION

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## SUMMARY

One of the major thrusts of the Air Force Armament Laboratory is to develop technologies for the advanced weapons required to accomplish "lethal defense suppression." In order to develop the long range investment strategy for this thrust, a Technology Base Definition Study for lethal defense suppression was conducted. A major product of this study was a Long-Range Integrated Technology Plan which: identifies specific technology efforts; outlines their required maturity dates to meet envisioned weapon system needs; and then establishes the investment strategy required to meet the thrust objectives. In addition to, and in support of the development of the long-range plan, a Threat Survey, a User Survey, a System Survey, and a Technology Survey, as well as a Cost and Operational Effectiveness Analysis, was conducted and published as separate reports. The study concluded that the required technologies to achieve a cost effective weapon system would best be directed toward a long-range, semi-ballistic missile with a dispenser/submunition warhead. Six major technology areas were described for further R&D exploitation through the year 1998 and a total of eleven elements were defined in the 12 year planning cycle.

## 1.0 PROGRAM OVERVIEW

Historically, air defense suppression has been accomplished by a combination of electronic countermeasures (ECM) and physical destruction. Very heavy emphasis on ECM has resulted in the incorporation of elaborate countermeasures (CM) by the air defense agencies. More recently, it has been concluded that lethal methods of suppressing enemy air defenses need additional emphasis, because of the fuller appreciation of the longer-term benefits of target destruction as compared to the transient nature of ECM. If a defended area were to be penetrated and attacked only once, objectives might be fully met by jamming the enemy's sensors; but if repeated penetrations of enemy air space are required, as in a protracted war, the attrition of this air defense resource through physical destruction has an ultimate payoff greater than by the repeated use of ECM alone. Combinations of lethal weapons, ECM, and tactics are anticipated to be needed for almost any operation involving penetration of air defense and these elements were considered in this study effort.

The Threat Definition has provided graphic descriptions of surface-to-air missiles, anti-aircraft artillery and laser weapon systems, illustrating the depth of technology being incorporated by the Soviets. Analysis of weapon operational characteristics and of their vulnerabilities was provided. Included in the threat are countermeasures to protect air defense systems for lethal suppression. Use is expected by countersuppression techniques, to include decoy emitters, fake signals, jammers, emission control, cooperative engagements between adjacent sites, and passive sensors.

Thousands of government laboratory technology tasks were reviewed during the Technology Survey, Reference (1). The most topically significant of these were documented with brief descriptions, key technologies, mission significance, status, schedule, and point of contact.

The Systems Survey reported, Reference (2), on both the U.S. and foreign lethal air defense suppression system developments. Some of these systems are dedicated to air defense suppression and others are more general purpose weapons, but have capabilities applicable to air defense suppression. The report included sections on Systems, Weapons, Platforms, C<sup>3</sup>I, and Target Acquisition Subsystems.

The User Survey, Reference (3), addressed current tactics, current capabilities, perceived future needs and analytical models. It was observed, for example, that the user tends to emphasize concern about near-term problems, rather than addressing the needs of long-range technology planning.

The objective of the COEA, Reference (4), was to assess and compare the merits of technologies for lethal defense suppression, and to provide supporting rationale for the development of specified technologies. Cost effectiveness was established by estimating the military worth of each target type and comparing the cost of defeating the targets with least cost mixes of inventory, developmental and advanced weapon concepts. The COEA compared the military value of five baseline weapon systems (GBU-15, MAVERICK-F, MRASM, JTACMS, and HARM) and a variety of potential defense suppression weapon development candidates (49 different types). Each weapon type was evaluated against twelve Soviet air defense threat units projected for the year 2000 including AAA sections, EW/GCI sites, nine different types of Soviet SAM batteries, and a tactical high energy laser/particle beam weapon.

The key analytical tool used for the COEA was the Armament Systems Utility Model (ASUM), a deterministic, expected-value computer model. Measures of effectiveness included cost per kill,

sorties saved, and aircraft saved. The COEA concluded that a long-range, semi-ballistic missile with dispenser/submunition warhead is the most cost effective weapon examined. To achieve this capability, growth in the following technologies is required: a. Target location sensors, both remote and self contained systems, carried on suppression aircraft, b. Aircraft avionics to provide precise and timely targeting data, c. Programmable inertial guidance units for midcourse guidance, d. Autonomous adverse weather seekers, e. High performance, fuel efficient propulsion units, and f. Improved cluster warheads with smart submunitions.

The Long-Range Technology Plan, Reference (5), provided time phased plans and roadmaps for technology research and development to assure the availability of the above improved technology for lethal air defense suppression in the 1995-2000 time frame. The various technologies which were selected for intensified development were considered in the context of a hypothetical, operating concept, to provide guidelines for making decisions in selecting components and choosing between competitive approaches. The individual technology plans were annotated with major events, action items, cost and risk estimates. The plans were developed to include three generations of tiers of activity, extending through the year 1998.

The remainder of this paper will summarize the features of the study by highlighting the task elements described above. Emphasis will be placed on rationalizing the decisions made on the technologies to be pursued through the long-range plan.

## 2.0 THE THREAT: DESIGN TRENDS IN SOVIET GROUND BASED AIR DEFENSE SYSTEMS

From the late 1940s to the present, the Soviet Union has steadily and at great cost and effort developed a sizeable and sophisticated ground based air defense network. In the interim, they have transitioned from a radar poor, fixed site, AAA, point defense concept to a systems rich, predominantly missile, dense and varied defense mixing both point defense of key assets and belted defense concepts. Rear area and territorial defenses maintain a mix of point defense and belted defense concepts. The forward mobile defense of Soviet/Warsaw Pact ground forces is built around a concept of dense, overlapping coverage by a variety of defensive systems employing varying acquisition and guidance modes. Both defense concepts are made possible by and derive from defensive weapon system capabilities mandated by Soviet design practices and priorities. The Soviets emerged from WWII with a defensive network that was insufficient to provide them with the desired level of air defense protection that they required relative to Western offensive aircraft. They have spent the period up to the present remedying that situation.

As a result of their efforts to close the relative gap between their defenses and Western offensive systems, the course of their progress has been marked by attention to certain specific development patterns. These patterns or priorities appear to remain viable today and will likely guide the development of future Soviet and Warsaw Pact ground based defenses in the future. Outstanding among these principles with regard to ground based air defenses are attention to mobility, mass numbers of systems, variation in systems, variation in guidance modes among systems, resistance to ECM, redundant coverage by differing systems, built-in operations security measures and storage of replaced systems so that inventories continually increase. These development and deployment practices continue to be followed and will likely always be part of Soviet air defense development practices.

In addition to their longstanding, basic practices, the Soviet development pattern has made it obvious that during certain periods they particularly direct and dedicate their efforts to overcome certain deficiencies or improve specific areas of their overall defense. Various references indicate that the Soviets are now concerned with a number of engagement and weapons systems parameters that have been exposed or are suspected weaknesses in their existing fielded defenses. These include limited multiple target engagement capability by individual weapon systems, emission control levels and practices, battery level target acquisition capabilities, defensive weapons minimum reaction times and deficiencies in low altitude engagement.

One method of closing off existing deficiencies in fielded systems that the Soviets have used in the past is to modify weapons in the field. This has been used to rapidly close off what were perceived as vital deficiencies in a manner without considerations of cost or effort. On other occasions, field modifications may be made at a more leisurely pace because the modification is technologically available and the original weapon system was initially fielded with the recognition that it would be later updated via modifications. Another method of closing off deficiencies is to concentrate on the development of remedial technologies for incorporation in follow-on systems. In recognizing these patterns, it is possible to see Soviet perceived deficiencies in one generation of weapons by the new capabilities which have been incorporated in the next generation.

### 2.1 MULTIPLE TARGET NEGEMENT CAPABILITY

In the first generation of Soviet defensive missile systems, both the lethality of the system and the number of airborne targets that could be engaged simultaneously were limited. With the deployment of follow-on systems, both deficiencies have been considerably reduced. With the exception of the smaller, lesser ranged and man portable weapons such as the SA-7, 9 and their follow-on replacements, the SA-13 and 14, the general trend in Soviet SAMs is in the direction of multiple engagement capabilities.

Both the SA-4 and the original version of the SA-6, the SA-6A, were limited to the simultaneous engagement of a single target by a battery. The limitation was imposed by the inability of the target acquisition radar to deal with more than one target at a time. This limitation was also true of all Soviet SAM systems which preceded the SA-4 and 6. The inability to engage more than one airbreathing threat at a time is a deficiency because it both restricts the level of attrition that can be imposed on the enemy per fielded SAM battery and also opens the battery to attack by saturation tactics from defense suppression aircraft.

## 2.2 POSSIBLE VULNERABILITIES

As the Soviets approach the 1990s, they confront the problem of maintaining or improving the quality of their air defense in comparison to their potential enemy's offensive weapon advances. Having spent much time and enormous resources in developing their mobile ground based defenses, they must maintain their capabilities and develop appropriate responses to Western attempts to utilize technology or numbers to bypass or overwhelm them. Existing deficiencies must be corrected. Future shortcomings must be recognized and developments begun on overcoming them. The challenging task undertaken is that of identifying vulnerabilities that are likely to remain in the 1990s and beyond that may be exploitable by future air defense suppression systems and tactics. Because the Soviets have developed and are projected to continue developing an integrated air defense network comprised of many different air defense systems, vulnerabilities of individual systems are largely compensated by other systems deployed in the same area.

The West has already begun development and in some cases deployment of new weapons which will challenge Warsaw Pact defenses in the 1990s. These include improved aircraft and missile performance specifically designed to operate at low altitude and high speed in all weather conditions. At higher altitudes, standoff weapons promise to have greater ranges, improved capabilities for launch and leave delivery, and greatly reduced CEPs. The same technological advancements which allow more accurate weapons delivery will contribute to more efficient, more accurate and increasingly responsive intelligence collection and reporting of targets. Improved tactical ECM for greater penetration assistance has been a longtime Western development effort which is now being field deployed. These are all low risk for expected Western weapon and intelligence developments, the need for which was long recognized and for which the Soviets were known to be developing countermeasures.

Of a more revolutionary nature, or at least new type threat, are Western deployments of advanced ballistic missiles (Pershing II), cruise missiles and the forecast development and deployment of stealth, or extremely low signature, attack and reconnaissance vehicles. These innovations combine, with a potentially greater Western emphasis in the future on attacking Warsaw Pact rear areas, as an outgrowth of U.S. development of the Airland Battle 2000 and Deep Attack concepts. Such Western doctrinal advances and the concurrent development of the means to effectively implement them will probably cause the Soviets to perceive a threat for which they are not currently prepared. NATO's past lack of sufficient, sophisticated, long-range threats to the Warsaw Pact's rear areas may change in the future. Such a development would no longer allow the Warsaw Pact to consider its rear areas sufficiently defended by the less well equipped, non-Soviet allies. Effectively, the area of the Warsaw Pact requiring a near total defense would be greatly increased.

Nevertheless, the Soviets are addressing major vulnerabilities that have existed in the past and are developing defensive counters to foreseen Western offensive advances. Examples include improvements in low altitude defense, ballistic missile attack, massed attack and ECM.

In a methodology for conducting future defense suppression efforts, certain specific technologies and principles appear to offer especially promising avenues for developmental and tactical concentrations of efforts. The combinations of basic physical properties, existing or emerging Western weapons technologies and apparent Soviet technology limitations appear to make particular areas key ones in which to play on future Soviet air defense deficiencies against defense suppression efforts. Particularly, they could play major roles in achieving the primary defense suppression methodology goal of defeating the key long-range Soviet/WP SAMs. These were the ideas and concepts reviewed and studied in this program.

## 3.0 SURVEYS AND PROGRAM METHODOLOGY

The program plan, shown in Figure 1, placed initial emphasis on the threat definition with briefings at the "all source" level followed by formal documentation. Team efforts began the data gathering phases of the Technology, Systems and User Surveys through personal contacts and literature searches. Drafts of these reports permitted work to be initiated on the Long-Range Technology Plan and the COEA, prior to final documentation of the earlier studies. These tasks were initiated early in the program and formed the basis for "Assessment of Capabilities Needed." (Quotes refer to blocks on Figure 1.) Tables 1 and 2 show the summary of the Technology and System Surveys.

As the surveys were being concluded, information was gathered and plans were prepared for the "Analysis Methodology" of the COEA. Computer models were selected, the programs were modified and input data was gathered. Study program personnel synthesized future weapon concepts based on hypothesized future capabilities, relative to the "Existing Lethal Systems" already known. These new concepts were studied; some were rejected, others were modified, and finally, a group of "Far-Term Generic Options" was selected with careful attention given to predictions of future technology. This group of concept options was further studies, with "Technology Projections" made for assigning values to those hardware and functional parameters which affect the COEA output. These values (e.g., seeker range and detection probability, weapon range and velocity, etc.) were subjected to intensive review and careful revision.

Several different figures of merit were used in determining the most cost-effective concepts which were then analyzed in greater depth. Analysis of the winning concepts determined which "Technology Needs" must be developed in order to fully realize the predicted future capability. From these technology needs, the outline of the top-level, long-range "Technology Roadmap" was generated, placing the different technologies in logical time sequence. Details were developed for the second and third-level roadmaps, conflicts were resolved, and narratives prepared.

A parallel path is noted in the top right of Figure 1. During the COEA, several current defense suppression weapons were selected to serve as baselines of comparison with the new generic concepts.

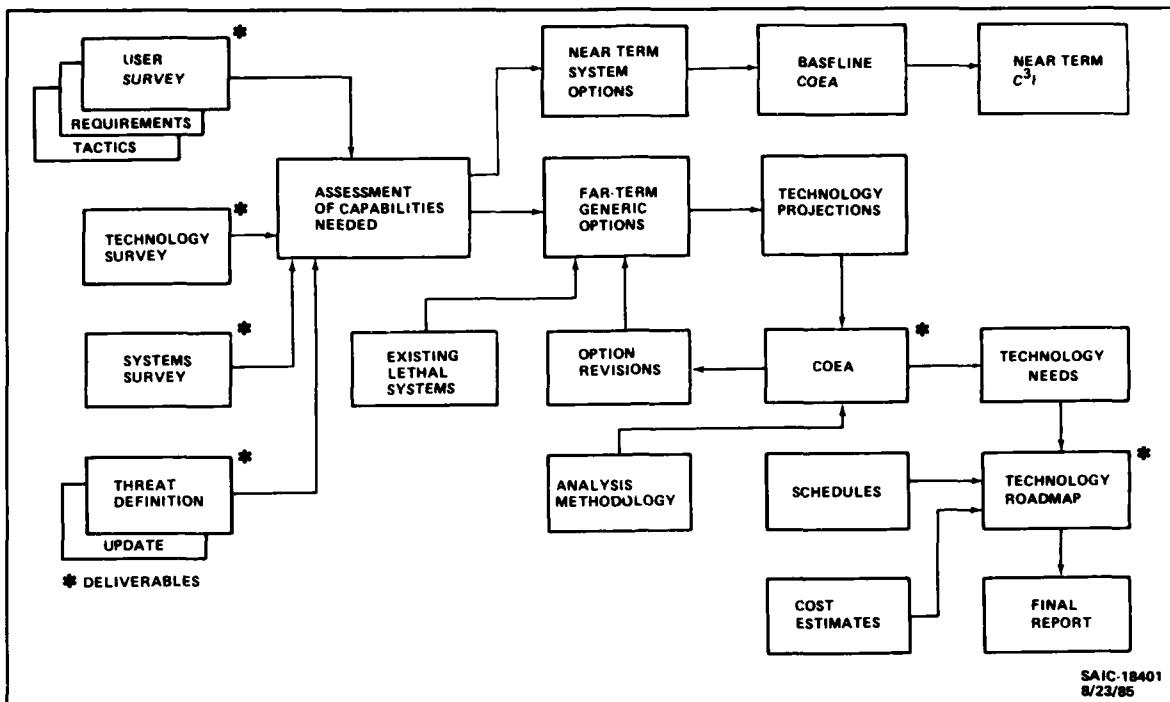


FIGURE 1. TECHNOLOGY PLANNING METHODOLOGY

The cost effectiveness of these baseline weapons and other data were used to determine the viability of near-term options for improvements in defense suppression.

**TABLE 1. TECHNOLOGY SURVEY OUTLINE**

- WEAPONS
  - AERODYNAMICS AND STRUCTURES
  - FLIGHT CONTROL
  - MID COURSE GUIDANCE
  - TERMINAL GUIDANCE
    - ▲ ELECTRO-OPTICAL
    - ▲ RADAR
    - ▲ MULTIMODE
  - ▲ SUPPORTING TECHNOLOGY
  - KILL MECHANISMS
  - COUNTERMEASURES
- AIRCRAFT WEAPONS INTEGRATION
  - CARRIAGE
  - AVIONICS SUPPORT
  - ELECTRICAL INTERFACE
  - MECHANICAL INTERFACE
- TARGET ACQUISITION
- MISCELLANEOUS TECHNOLOGIES

TABLE 2. SUMMARY OF SYSTEMS SURVEY

- INCLUDES:
  - 21 WEAPONS
  - 15 PLATFORMS
  - 10 C<sup>3</sup> SYSTEMS
  - 11 TARGET ACQUISITIONS
- WEAPONS DATA INCLUDES:
  - CONFIGURATIONS
  - COMPONENT LAYOUTS
  - PERFORMANCE ENVELOPES
  - PHYSICAL PARAMETERS
  - STATUS

#### 4.0 COEA SUMMARY

The objective of the COEA was to assess and compare the merits of technologies for lethal defense suppression, and to provide supporting rationale for the development of specified technologies. Cost effectiveness was established by estimating the military worth of each target type and comparing the cost of defeating the targets with least cost mixes of inventory, developmental and advanced weapon concepts.

The analysis was designed to critically compare weapon system technologies and operational considerations to the prioritized target list generated in the Threat Definition. Input data were based on the information obtained in an extensive Technology Survey and the Systems Survey. The methodology included operational tactics and strategy identified in the User Survey.

#### 4.1 ASSUMPTIONS

The COEA analyzed the utility of engaging enemy air defense target elements with a variety of baseline and conceptual lethal weapon systems. The effort was guided by the following assumptions:

- The governing scenario was for a Central European conflict in the late 1990s to early 2000s time frame with the projected enemy technology for target acquisition, threat/weapon systems, C3, delivery platforms and ECM.
- The friendly weapon launch platforms were typified by projected upgrades to existing F-16 and F-4G aircraft. The force will also include F-15E, FA-18 and Advanced Tactical Fighter (ATF) aircraft which are expected to be integrated into the NATO defense forces during this period.
- Available support systems which were assumed to be operational during the conflict included:
  - Joint Surveillance and Target Attack Radar System (JSTARS)
  - Precision Location Strike System (PLSS)
  - Military Satellite Targeting System (MILSTAR)
  - Global Positioning System (GPS)
  - Joint Tactical Information Distribution System (JTIDS)
  - Joint Fusion Center (JFC)

#### 4.2 ANALYSIS

The analysis compared the military value of five baseline weapon systems (GBU-15, MAVERICK-F, MRASM, JTACMS and HARM) and a variety of (49 different types) potential defense suppression weapon development candidates. Each weapon was evaluated against twelve Soviet air defense threat units projected for the year 2000, including AAA sections, EW/GCI sites, nine different Soviet SAM batteries and a tactical high energy laser/particle beam weapon.

The Soviet AD systems were given a relative worth in order to determine and quantify their value in battle. Figure 2 shows the systems considered in the simulation. The scenarios that were studied via the simulation model are shown in Figure 3 indicating the baseline weapons, candidate seekers, sortie depth of penetration, warhead weight and stores per aircraft.

The results have provided added support for the conventional wisdom that aircraft survival and least-cost weapon mixes require long standoff to permit launch from outside the lethal range of the threat. They have also reinforced the strategy of attacking air defense units in close air support zones (>30 km) and tactical air interdiction zones (30-150 km), on a priority basis, to take advantage of weapon standoff and to reduce ingress and egress attrition prior to operation against deep interdiction (<150 km) targets. This defense rollback tactic may not be practical for high-priority fixed targets, but is important to cost-reduction for the lethal air defense suppression mission.

Some of the other findings of the COEA were not so obvious, but may be more important to weapon development and acquisition decisions. For example, consider Figure 4, which compares the three best baseline weapons (MAVERICK at 30 km, HARM at 80 km and JTACMS at 200 km). The summary represents the resources to kill 90 percent of the Warsaw Pact target ADUs, using the optimum mix of weapons per mission to obtain the desired kill level. In the postulated Central European Scenario, additional sorties may be critical to the overall ground-battle outcome and events may dictate that all available assets will have to be committed to stop the enemy's armored vehicle attacks. The tanks must be stopped to prevent the Warsaw Pact armored divisions from rolling across Western Europe in only a few days according to their expected battle plan. Only tactical air forces can provide the concentration of firepower and battlefield mobility needed for this mission. This means that lethal defense suppression sorties would be limited in favor of Close Air Support.

SYSTEM	RELATIVE TARGET VALUE ESTIMATE	SOVIET SYSTEM COST ESTIMATE	CLOSEST U. S. EQUIVALENT
1. SA-X-12 BATTERY	42	\$100M	PATRIOT
2. TACTICAL HEL	29	60M	HEL
3. SA-10 BATTERY	21	28M	IMPROVED HAWK
4. SA-11 BATTERY	19	28M	IMPROVED HAWK
5. EW/GCI SITE	15	30M	NOT ESTABLISHED
6. SA-6 BATTERY	15	17M	HAWK
7. SA-4 BATTERY	7	10M	NIKE HERCULES
8. SA-8 BATTERY	6	42M	ROLAND
9. SA-13 BATTERY	5	10M	NO COUNTERPART
10. SA-3 SITE	4	10M	NO COUNTERPART
11. SA-9 PLATOON	3	5M	CHAPARRAL
12. ZSU-23/4 SECTION	2	9M	SGT. YORK

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FIGURE 2. RELATIVE WORTH OF AIR DEFENSE SYSTEMS

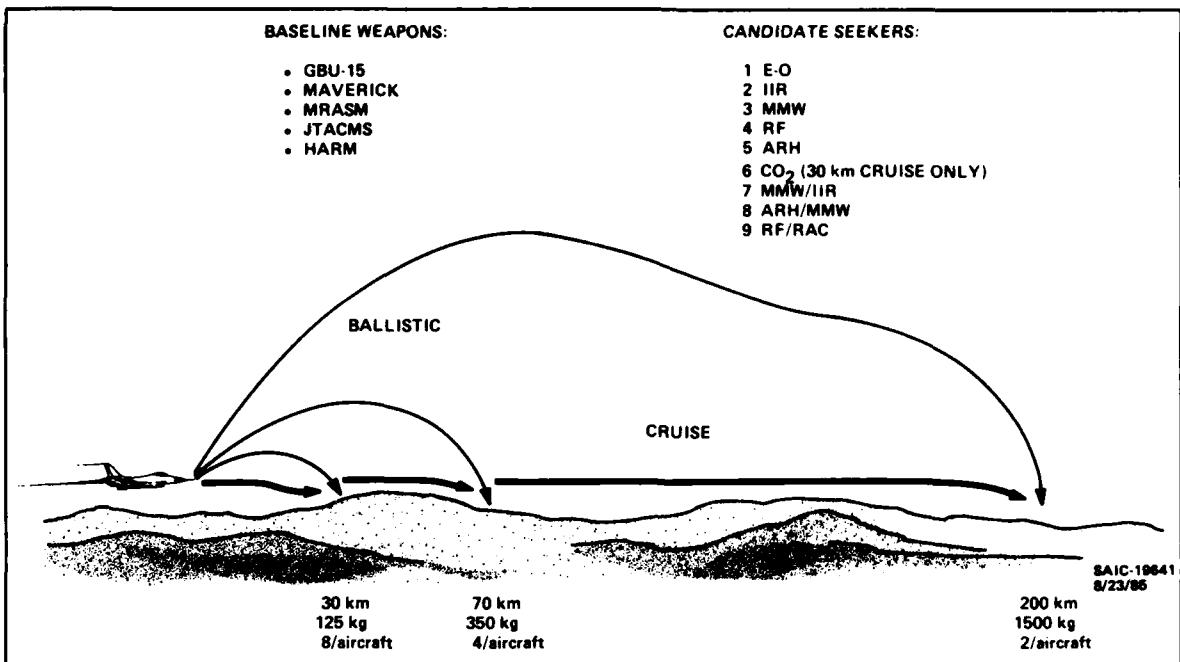


FIGURE 3. CANDIDATE WEAPON CONCEPTS

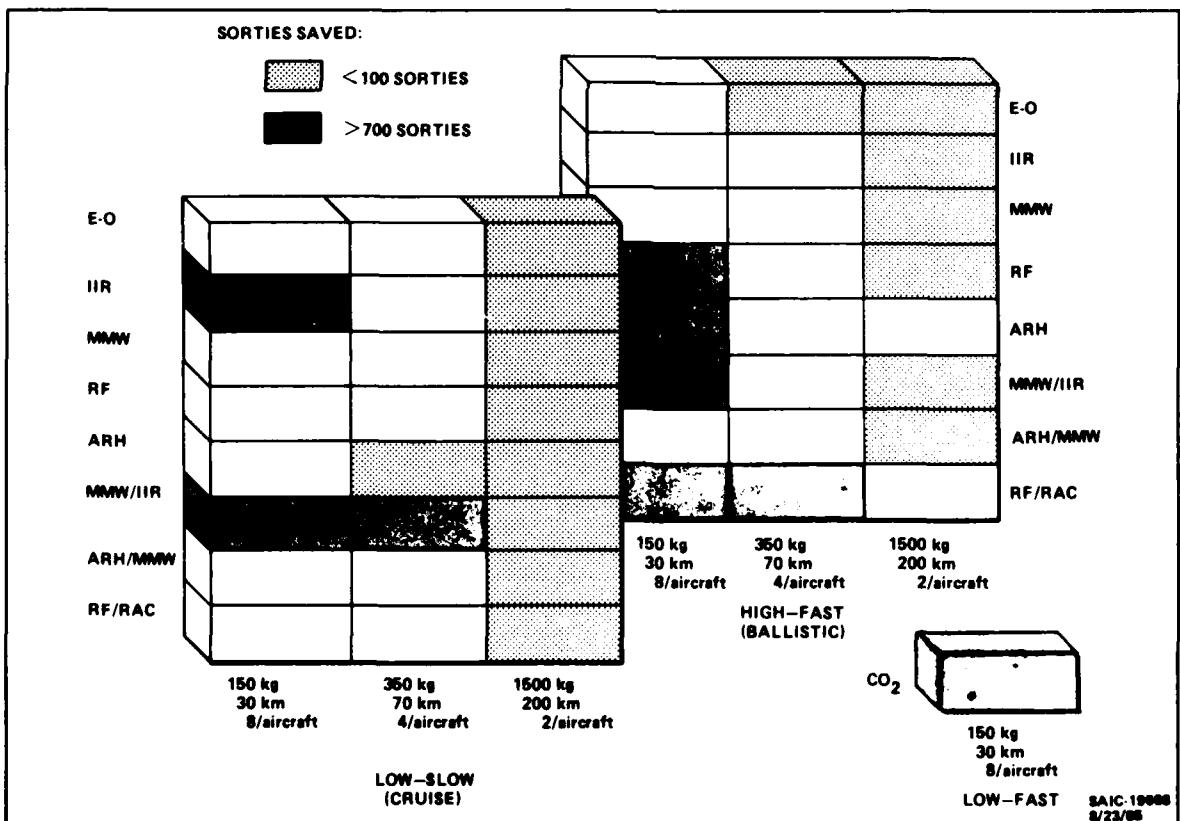


FIGURE 4. CANDIDATE CONCEPT COMPARISON

Thus in a longer war, standoff range and aircraft survival may be paramount to cost reduction, but if the friendly air bases may be overrun within a few days, then individual sortie effectiveness (kills per sortie) is ultimately more important to the outcome of the conflict than total aircraft losses or costs per kill.

The same logic applies when comparing weapons with and without night/all-weather operational capability. If the available sorties are severely limited by time, as well as numbers of aircraft, attrition and cost are of secondary importance and sortie effectiveness is primary in terms of affecting the final outcome of the ground-war. Night/all-weather capability can provide a major increase in sorties available, as well as an increase in average sortie effectiveness.

#### 4.3 RESULTS

In summary, study results have been based on three figures of merit, rather than cost alone. These measures are:

- a. Number of sorties saved;
- b. Number of aircraft saved; and
- c. Cost saved;

by adding a weapon concept variant to the inventory of baseline weapons. Resources are saved because the added concept is more effective than the baseline weapons.

A summary of values determined for 49 variants considered is in Figures 5 and 6. Note that the variants are identified by terminal guidance, (EO, IIR, etc.), gross weight (125, 250, and 1500 KG), and as either ballistic or cruise missiles. The weights correspond to ranges of 30, 70 and 200 km respectively.

From the figure (Ballistic Concepts), note that the RF/RAC variant excels in every category except one (cost saved, 1500 KG version). The characteristics that made this variant a winner are all-weather, autonomous operation with an accurate terminal seeker, improved propellant technology for greater range/payload ratios, advanced warhead lethal mechanism technology and optimal fusing techniques. Other variants with these characteristics would be comparably effective.

From Figure 4, the CO<sub>2</sub> laser guided variant is outstanding in sorties and cost saved, but is not recommended because of the projected cost of the associated pods and the operational restrictions imposed by the Command-to-Line-of-Sight (CLOS) guidance and other reservations. Hence the IIR seeker approach is preferred in the cruise missile category where seeker range and all-weather capability are less critical. This is shown in Figure 7 as the second best concept.

The selection of a dual-mode, autonomous RF seeker and a 200 km standoff range as the most cost effective variant is based on the necessity for countering the higher value, longer range targets. Thus the longest range ballistic variant is the recommended approach. Technologies required include RF/RAC terminal guidance, low cost inertial guidance, ducted rocket propulsion, and a cluster munition such as SFW and SADARM which is optimized for light materiel targets (i.e., SAM sites and mobile ADU batteries) summarized in Figure 8. These are the key technologies recommended for the technology roadmaps.

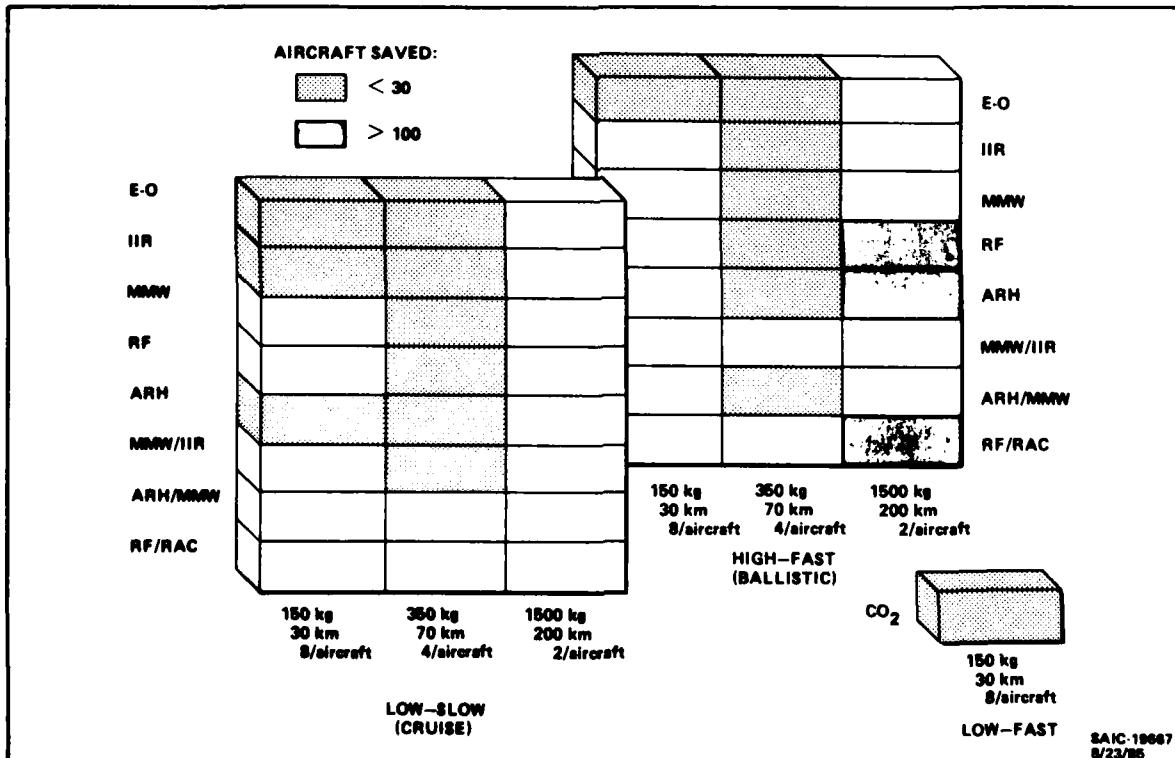


FIGURE 5. CANDIDATE CONCEPT COMPARISON

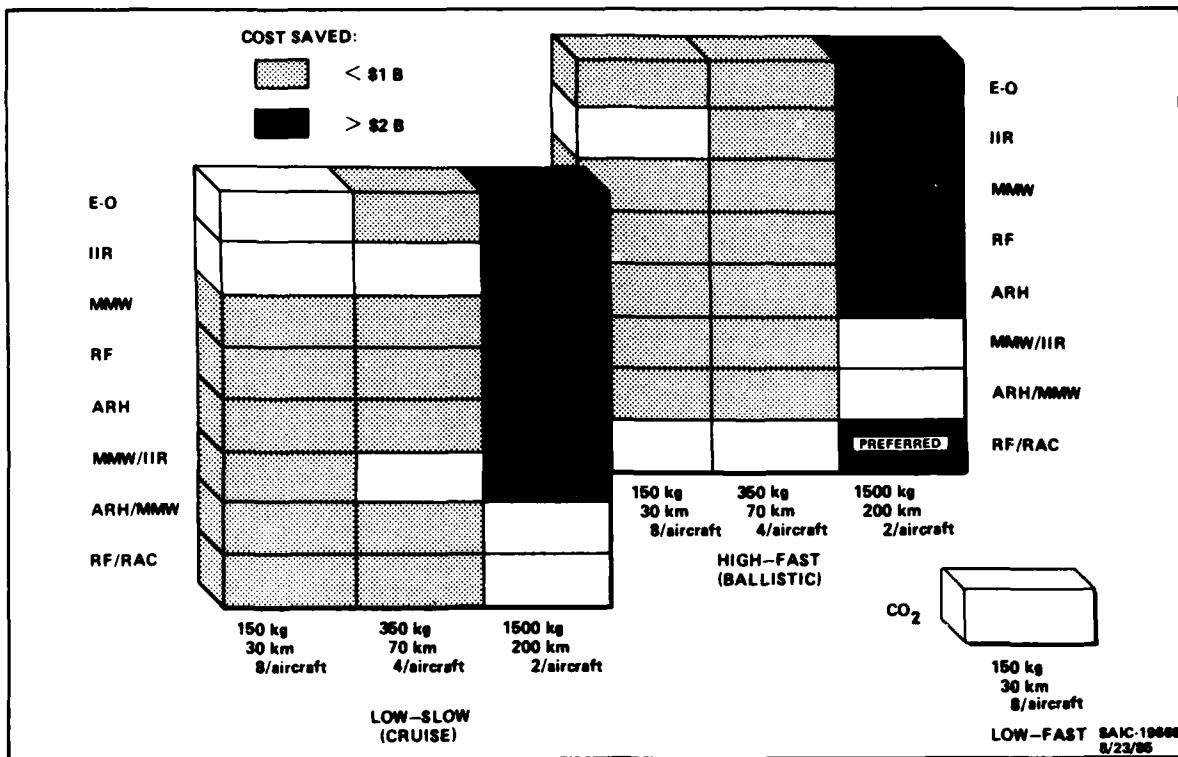


FIGURE 6. CANDIDATE CONCEPT COMPARISON

- MISSILE – 2 STAGE SOLID/SOLID DUCTED ROCKET
- ENGAGEMENT RANGE – MIN 300 FT, MAX 30km
- TARGET ASPECTS –  $\pm 180^\circ$ ,  $90^\circ$  SEEKER GIMBAL LIMIT
- LAUNCH SPEEDS – MIN .5 MACH, MAX 2.5 MACH
- TARGETING – RHAW, RADAR, VTAS, LASER RANGER
- CUEING – INFLIGHT INERTIAL UPDATE VIA MIL-1760
- WARHEAD – 24 kg, TUNGSTEN FRAGS, AIMABLE
- FUZING – RESETTABLE PRIOR TO LAUNCH, RF/IR LASER
- GUIDANCE – INS MIDCOURSE, IIR TERMINAL
- SURVIVABLE

SAIC-21301/2916  
8/23/85FIGURE 7. SHORT RANGE WEAPON CHARACTERISTICS (8/AC)  
SECOND BEST

- MISSILE – 2 STAGE SOLID/DUCTED ROCKET
- ENGAGEMENT RANGE – MIN 10 km, MAX 200 km
- TARGET ASPECTS –  $\pm 60^\circ$ ,  $30^\circ$  SEEKER GIMBAL LIMIT
- LAUNCH SPEEDS – MIN .5 MACH, MAX 2.5 MACH
- CUEING – INFLIGHT INERTIAL UPDATE VIA MIL-1760
- WARHEAD – SFW, DISPENSER
- FUZING – IR SENSOR
- GUIDANCE – INS MIDCOURSE, RF/RAC
- TARGETING – RHAW, RADAR, PLSS, JSTARS, GPS

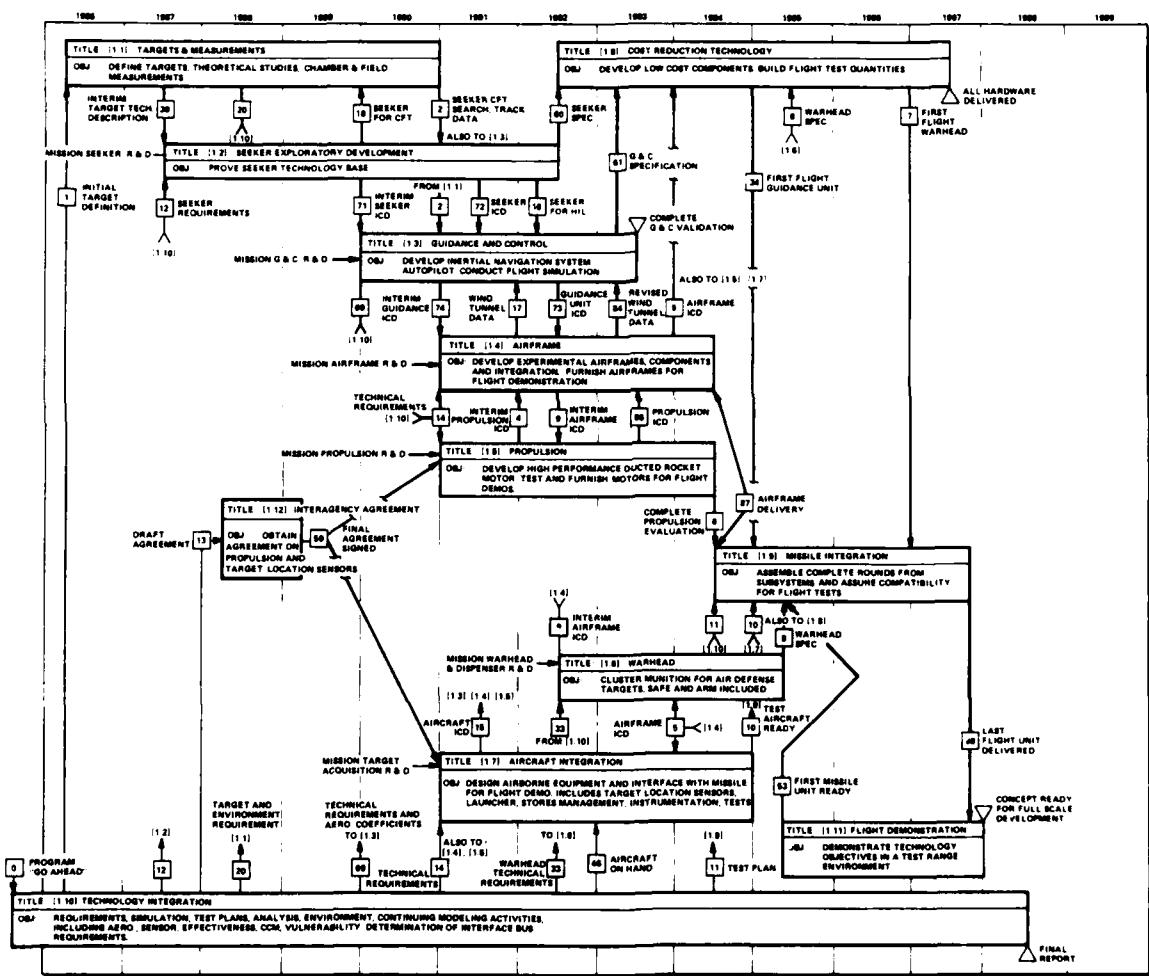
SAIC-21302/2917

FIGURE 8. LONG RANGE WEAPON CHARACTERISTICS (2/AC)  
BEST WEAPON

## 5.0 LONG-RANGE TECHNOLOGY PLAN OVERVIEW

Following the COEA findings, the selected concept was first restudied in more detail. The technology survey and other documentation was then reviewed for assessing applicable tasks. To determine a logical sequence for conducting tasks, technology interrelationships were investigated, and this led to the conclusion that early emphasis must be placed on those technologies with highest risk and the greatest payoff in terms of new defense suppression capabilities. High-risk/high-payoff technologies (in order of payoff) have been judged to be the seeker, followed by the inertial guidance system, airframe, and warhead. Parallel development of propulsion and target location sensors is required by other agencies to complement these efforts.

The next activity was preparing an overview roadmap, shown in Figure 9, and adjusting sequences and schedules among the different technologies. This was followed by the second level maps and designation of the events and actions which interconnect the various roadmap activities. Iterations of the schedule took place to resolve incompatibilities. The narratives were then prepared to list the technologies included to document the scope of the individual tasks. The resultant Long-Range Technology Plan was designed to support more than one weapon concept with full-scale development beginning around 1998. Actually, partial completion of some of these tasks can "spin off" earlier weapon system developmental projects, if desired. For example, once the seeker and inertial guidance concepts have been proven, they could be adapted to an existing airframe, without awaiting propulsion or other improvements called for in the complete weapon development cycle.



From the COEA results, highest priority development is a long-range (nominal 200 km) air launched missile with inertial midcourse guidance, active radar terminal guidance and a dispersed smart submunition warhead.

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**MIDCOURSE GUIDANCE FOR THE  
U. S. ARMY'S LOW COST FIBER OPTIC GUIDED MISSILE**

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INTRODUCTION

The use of fiberoptics as a datalink for a standoff missile offers unique system capabilities heretofore not achievable with conventional systems. It allows out of line of sight target acquisition, missile guidance and control, remote autotracker engagement, and precision guidance to impact. Guidance is to be implemented in a remote hidden gunners station. The high bandwidth fiber link allows bi-directional simultaneous transmission of missile imaging sensor video and on board sensor signals to the ground while missile steering commands are sent to the missile. The indirect fire capability using the countermeasure resistant datalink allows high system survivability and effectiveness.

The U. S. Army Missile Command's fiberoptic guided missile program (FOG-M) is demonstrating this technological capability for a standoff tactical missile application. The capability of reusing guidance and control hardware that remains in a protected gunners station allows the system acquisition and operational costs to be considerably reduced. Implementation of the midcourse guidance function in ground computers allow more powerful G&C techniques to be utilized than in a space constrained missile airframe. A method of midcourse guidance that takes advantage of these capabilities is discussed below.

FOG-M GUIDANCE AND CONTROL SCHEME

Presently the fiberoptic guided missile uses a combination of guidance and control techniques throughout its mission profile. During the vertical launch boost phase (approx. 2.8 seconds), the missile is stabilized in the body axis frame using 3 single axis body rate gyros, or a combination of a single axis roll gyro and use of body rate information derived through differentiating the seeker gimbal angles and adding the angular rates from pitch and yaw gyros mounted on the imaging seeker. During this maneuver, a pitchover command is given to the actuators through the fiberoptic datalink. The missile altimeter reading is continuously compared to the altitude commands to establish when the missile has reached the designated cruise altitude. After this time the derived body inertial attitude in the pitch plane is also used to dampen the altitude loop.

After suitable filtering, the steady state altitude is reached, and a flag is set in the gunners station computer software to the second guidance mode, cruise guidance. During this mode, the missile uses attitude stabilization from the integrated rate gyro signals to maintain missile heading. This form of guidance is sensitive to crosswinds, and large lateral excursions can result over the mission flight time if additional corrections to flight path are not made. These corrections are currently made by manual commands from the system gunner using a force button controller, by use of a digital correlator to sense lateral displacement in the video obtained from the flight imaging seeker, or by signals from an automatic tracker that has been locked onto a downrange object along the azimuth to the desired target area. Details of this guidance system are described in reference [1].

The present guidance methods are adequate if the gunner can remain a part of the guidance loop during the time required to make corrections or lock the autotracker on the "target". However, if the gunner is unable to be a part of the loop during the entire cruise phase, the necessary corrections will not be made. This situation is likely if the gunner is required to be servicing other missiles already in the target area for the target selection functions. Another reason an alternate form of midcourse guidance is desired is for guidance of individual missiles which, while being guided simultaneously, have different mission profiles (e.g. flying around hills or up valleys).

A form of strapdown inertial guidance could adequately do this function, but has the disadvantage of requiring more expensive sensors on each missile. If a method can be devised to allow ground based equipment to perform the midcourse guidance function, there will be savings on system acquisition costs, since the equipment can be used repeatedly. The information on missile position can also be used as a target acquisition aid for additional targets discovered during the flyout by observing the sensor video.

This paper presents an analysis of a method for achieving the midcourse guidance using a digital correlator and known "white rocks", or positions of objects that will appear in the seeker video. A variety of filtering techniques and filter mechanizations are analyzed and the results presented in the following sections. Algorithms are developed which use updates from the ensemble of white rocks or waypoints to correct the missile attitude (heading, primarily) at each waypoint and predict time or range to go to the target area. These estimators of navigation parameters have no effect on missile control other than a few guidance corrections, and can work with either a strapdown or gimballed seeker tracking the "white rock". If the coordinates of the "white rocks" are not known, then the estimator can still accurately determine heading and wind corrections.

In the present work models for the system dynamics and the measurement process have been formulated and six extended Kalman filter designs have been developed and tested for state estimation accuracy and efficiency. Work has also been performed to improve the efficiencies of the algorithms so that the estimates are available in a timely manner for midcourse navigation and to aid the operator.

#### SYSTEM AND MEASUREMENT MODELS

This section describes the underlying system models, measurement models, and filter algorithms. The equations of motion for a six degree of freedom rigid body are given. Various methods of parameterizing the direction cosine matrix are described. Three methods are presented: Euler angles, Euler parameters and Rodrigues parameters. Attention is focused on the relations between measured quantities and the state variables. This section concludes with the formulation of the nonlinear estimation problem and with typical results of the linearization about the current state estimates for both the system and measurement equations.

#### Equations of Motion

For the purposes of this study the missile is modeled as a rigid body. Its motion, neglecting earth's rotation, is governed by the following differential equations:

$$\begin{bmatrix} \dot{x} \\ \dot{y} \\ \dot{z} \end{bmatrix} = I_C^B \begin{bmatrix} u \\ v \\ w \end{bmatrix}, \quad (\text{Kinematics}) \quad (1)$$

$$\begin{bmatrix} \dot{u} \\ \dot{v} \\ \dot{w} \end{bmatrix} = -I_\Omega^B \begin{bmatrix} u \\ v \\ w \end{bmatrix} + g \frac{B_C I}{m} + \frac{F}{m}, \quad (\text{Dynamics}) \quad (2)$$

$$\begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{bmatrix} = \bar{T}^{-1} \left[ -I_\Omega^B \bar{T} I_\omega^B + \bar{T} \right], \quad (\text{Dynamics}) \quad (3)$$

$$\dot{s} = S \bar{\epsilon} \quad (\text{Kinematics}) \quad (4)$$

where

$\begin{Bmatrix} x \\ y \\ z \end{Bmatrix}$  = coordinates of the center of mass in the inertial coordinate system with origin at the target.

$\begin{Bmatrix} u \\ v \\ w \end{Bmatrix}$  = components of the center of mass velocity in the body coordinate system,

$I_C^B$  = transformation matrix (from body to inertia),

$I_\omega^B$  = angular velocity vector of the body in the inertial frame of reference.

$\left. \begin{matrix} p \\ q \\ r \end{matrix} \right\}$  = components of  $\underline{\omega}^B$  in the body-fixed frame,

$\underline{\Omega}^B$  =  $\begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix}$  = angular velocity matrix for the body in the inertial frame of reference,

$\underline{B}_C^I$  =  $\begin{bmatrix} B_{C1}^I \\ B_{C2}^I \\ B_{C3}^I \end{bmatrix}$ ,

$m$  = mass of the body.

$g$  = acceleration of gravity.

$\underline{I}$  =  $\begin{bmatrix} I_{xx} & 0 & 0 \\ 0 & I_{yy} & 0 \\ 0 & 0 & I_{zz} \end{bmatrix}$ ,

$I_{xx}$ ,  $I_{yy}$ ,  $I_{zz}$  = principal moments of inertia of the body about its mass center,

$\underline{\omega}$  = vector expressed in terms of the angular velocity of the body in the inertial frame of reference,

$\underline{s}$  = vector parameterizing the transformation matrix  $I_C^B$ ,

$S$  = matrix relating the body angular velocity to the time rate of change of vector  $\underline{s}$ ,

$\underline{F}$  = nongravitational external forces acting on the missile,

$\underline{T}$  = external moments about the mass center.

#### Parameterization of the Direction Cosine Matrix

Three different parameterizations of the direction cosine matrix  $I_C^B$  have been used: Euler angles, Euler parameters, and Rodrigues parameters.

#### Euler Angles Formulation

The direction cosine matrix  $I_C^B$  can be expressed in terms of Euler angles as follows.

Let  $\underline{s}^T = [\varphi, \theta, \psi]$

where  $\varphi$  = roll angle

$\theta$  = pitch angle

$\psi$  = yaw angle.

Then<sup>[2]</sup>

$$I_C^B = \begin{bmatrix} c\varphi c\theta & -s\varphi c\theta + c\varphi s\theta s\psi & s\varphi s\psi + c\varphi s\theta c\psi \\ s\varphi c\theta & c\varphi c\psi + s\varphi s\theta s\psi & -c\varphi s\psi + s\varphi s\theta c\psi \\ -s\theta & c\theta s\psi & c\theta c\psi \end{bmatrix} \quad (5)$$

where  $c.$  =  $\cos (\cdot)$

$s.$  =  $\sin (\cdot)$ .

$$S = \begin{bmatrix} 1 & \sin\varphi \tan\theta & \cos\varphi \tan\theta \\ 0 & \cos\varphi & -\sin\varphi \\ 0 & \sin\varphi \sec\theta & \cos\varphi \sec\theta \end{bmatrix} \quad (6)$$

### Euler Parameters Formulation

In this case, vector  $\underline{s}$  is a 4-vector

$$\underline{s}^T = [\epsilon_1, \epsilon_2, \epsilon_3, \epsilon_4], \quad \epsilon_1^2 + \epsilon_2^2 + \epsilon_3^2 + \epsilon_4^2 = 1.$$

It can be shown [3] that

$$I_{CB} = \begin{bmatrix} 1 - 2\epsilon_2^2 - 2\epsilon_3^2 & 2(\epsilon_1\epsilon_2 - \epsilon_3\epsilon_4) & 2(\epsilon_1\epsilon_3 + \epsilon_2\epsilon_4) \\ 2(\epsilon_1\epsilon_2 + \epsilon_3\epsilon_4) & 1 - 2\epsilon_1^2 - 2\epsilon_3^2 & 2(\epsilon_2\epsilon_3 - \epsilon_1\epsilon_4) \\ 2(\epsilon_1\epsilon_3 - \epsilon_2\epsilon_4) & 2(\epsilon_2\epsilon_3 + \epsilon_1\epsilon_4) & 1 - 2\epsilon_1^2 - 2\epsilon_2^2 \end{bmatrix} \quad (7)$$

$$S = \frac{1}{2}E = \frac{1}{2} \begin{bmatrix} \epsilon_4 & -\epsilon_3 & \epsilon_2 & \epsilon_1 \\ \epsilon_3 & \epsilon_4 & -\epsilon_1 & \epsilon_2 \\ -\epsilon_2 & \epsilon_1 & \epsilon_4 & \epsilon_3 \\ -\epsilon_1 & -\epsilon_2 & -\epsilon_3 & \epsilon_4 \end{bmatrix} \quad (8)$$

### Rodrigues' Parameter Formulation [3]

Let

$\underline{s}^T = [\rho_1, \rho_2, \rho_3]$ , where  $\rho_1, \rho_2, \rho_3$  are the Rodrigues parameters. The direction cosine matrix becomes

$$I_{CB} = \frac{\begin{bmatrix} 1 + \rho_1^2 - \rho_2^2 - \rho_3^2 & 2(\rho_1\rho_2 - \rho_3) & 2(\rho_3\rho_1 + \rho_2) \\ 2(\rho_1\rho_2 + \rho_3) & 1 + \rho_2^2 - \rho_3^2 - \rho_1^2 & 2(\rho_2\rho_3 - \rho_1) \\ 2(\rho_3\rho_1 - \rho_2) & 2(\rho_2\rho_3 + \rho_1) & 1 + \rho_3^2 - \rho_1^2 - \rho_2^2 \end{bmatrix}}{(1 + \rho_1^2 + \rho_2^2 + \rho_3^2)} \quad (9)$$

and the matrix  $S$

$$S = \frac{1}{2}[I - R + \underline{s}\underline{s}^T] \quad (10)$$

where  $I$  is the  $3 \times 3$  identity matrix,

$$\text{and } R = \begin{bmatrix} 0 & \rho_3 & -\rho_2 \\ -\rho_3 & 0 & \rho_1 \\ \rho_2 & -\rho_1 & 0 \end{bmatrix} \quad (11)$$

The relative efficiency of the three parameterization schemes has been studied by solving a simple rigid body kinematics problem: for prescribed angular velocity  $(p(t), q(t), r(t))$ ,  $t \geq t_0$  and initial conditions at  $t = t_0$ , find the orientation of the body at time  $t \geq t_0$ . This amounts to solving Eqn. (4) subject to given initial conditions. We have selected

$$p = 0.2 \sin(4\pi t), \quad q = 0.2 \sin(2\pi t), \quad r = 0.1.$$

Table 1 lists a measure of the relative efficiency (RE) of the three formulations. A fixed step 4th order Runge-Kutta method was employed for the numerical integration of Eqn. (4). The same step size was used in all computations, because the right hand sides of the differential equations are sinusoidal with the frequencies determined by  $p, q, r$ .

TABLE 1

Parameterization Method	Number of Differential Eqns. Solved	RE = $\frac{\text{Run Time for a Method}}{\text{Run Time for Euler Angles Formulation}}$
Euler Angles	3	1.00
Euler Parameters (Quaternion Method)	4	0.79
Rodrigues Parameters	3	0.72

## Measurement Equations

A number of onboard sensors provide measurements which are, in general, nonlinear functions of the state variables. The sensors considered in the various filter designs presented in this paper are summarized below:

Seeker potentiometers: provide seeker gimbal angles relative to the missile body; pitch ( $\theta_G$ ) and yaw ( $\psi_G$ ).

Seeker gyros: provide inertial rates of the line-of-sight in seeker axes; pitch ( $\dot{\theta}_p$ ) and yaw ( $\dot{\theta}_y$ ).

Altimeter: supplies the altitude above mean sea level ( $h_{MSL}$ ).

Missile gyros: provide the inertial rates of the body in body axes, p, q, r.

TV correlator: provides the "white rock" angles  $\alpha$  and  $\beta$  in pixel space.

The following equations relate the measured quantities to state variables in the ideal case that sensors do not introduce any type of measurement errors.

$$\theta_G = \tan^{-1} \left( \frac{B_{R_3}}{B_{R_1}} \right), \quad (12)$$

$$\psi_G = \tan^{-1} \left[ \frac{B_{R_2}}{\left( \left( B_{R_1} \right)^2 + \left( B_{R_3} \right)^2 \right)^{\frac{1}{2}}} \right] \quad (13)$$

$$\dot{\theta}_p = \frac{z\dot{x} - x\dot{z}}{x^2 + y^2 + z^2}, \quad (14)$$

$$\dot{\theta}_y = \frac{x\dot{y} - y\dot{x}}{x^2 + y^2 + z^2}, \quad (15)$$

$$\alpha = \sin^{-1} \left[ \frac{c_{31}x + c_{32}y + c_{33}z}{(x^2 + y^2 + z^2)^{\frac{1}{2}}} \right], \quad (16)$$

$$\beta = \tan^{-1} \left[ \frac{c_{21}x + c_{22}y + c_{23}z}{c_{11}x + c_{12}y + c_{13}z} \right], \quad (17)$$

$$h_{MSL} = -z + \text{constant}. \quad (18)$$

where

$$\begin{bmatrix} B_{R_1} \\ B_{R_2} \\ B_{R_3} \end{bmatrix} = B_C^I \begin{bmatrix} x \\ y \\ z \end{bmatrix}, \quad \text{and } c_{ij} = B_{C^I_{ij}}.$$

### System Model

The force,  $\underline{F}$ , in equation (2) is the sum of the aerodynamic and propulsive forces. In the present work, the force,  $\underline{F}$ , has been modeled as the sum of a deterministic aero-control force,  $\underline{F}_D$ , and a random force,  $\underline{F}_R$ , due to wind disturbances and propulsion system imperfections. In the numerical experiments presented herein, the deterministic force,  $\underline{F}_D$ , has been chosen to be

$$\underline{F}_D = -mg \frac{B_C I}{3}$$

(i.e. the nominal flight path is a well controlled straight line in the absence of random disturbances) and the random force has been modeled as

$$\underline{F}_R = \underline{F}_{BIAS} + \underline{\delta F}$$

where  $\underline{F}_{BIAS}$  is primarily due to wind biases and  $\underline{F}$  is primarily due to wind fluctuations and system imperfections. The wind fluctuations have been modeled as white Gaussian process with zero mean.

Similar statements are implied for the aero-control torques,  $\underline{T}$ , in equation (3).

For the filters that take into account all the kinematical and dynamical equations (filters 1A, 1B, 1C) the mean external moment about the center of mass has been taken equal to zero. On the other hand, in the filters where the angular rates are assumed prescribed (filters 2A, 2B, 2C),  $p$ ,  $q$ ,  $r$  have been modeled as zero-mean random Gaussian sequences so that equation (3) is no longer needed.

In all cases, the measurement biases have been included as states.

### Measurement Models

Equations (12) - (18) are valid in the ideal case that the sensors do not introduce any errors. Herein, each sensor is assumed to produce measurements which are contaminated by a random bias and white Gaussian noises. Thus, these bias and zero mean terms are added to equations (12) - (18) to produce the measurement models. The random bias is considered a random constant, i.e. it has a constant value during a flight but it varies from flight to flight. Mathematically, this is expressed by assigning random initial conditions to a differential equation of the form  $\dot{b} = 0$ .

### The Estimation Problem

Equations (1) - (4) are nonlinear stochastic differential equations with additive random noise. The output of these differential equations is a Markov vector process. The problem considered here is that of estimating the state vector,  $\underline{x}(t)$ , based on a number of nonlinear observations contaminated by additive noise. The problem is solved as one with continuous dynamics and discrete-time measurements. A linearized Kalman filter and several extended Kalman filters have been implemented and their performance examined for various levels of system disturbances and measurement noise. Some higher order filters have been considered for the nonlinear filtering problem described above. Among them the Iterated Extended Kalman filter [4], the Gaussian second order filter [4, 5], and the maximum a posteriori (MAP) filter [6]. Due to the dramatic increase in the computational burden required in implementing the higher order filters and the satisfactory results obtained with the Extended Kalman filter, the higher order filter algorithms were not considered serious candidates for this application.

### Kalman Filters

The estimation problem is nonlinear. Before applying the linear estimation theory one must linearize the governing differential equations of the system dynamics and the nonlinear algebraic equations describing the measurement process. In this application of flying straight-and-level segments from waypoint to waypoint the nominal motion is extremely simple.

### Linearized Kalman Filter

In order to linearize the dynamical equations one assumes a reference trajectory  $\underline{x}(t)$  and defines the state deviations from the reference trajectory

$$\dot{\underline{x}} = \underline{x} - \dot{\underline{x}}.$$

Similarly, in order to linearize the measurement equations, one defines the reference measurements

$$\dot{\underline{z}}_k = \underline{h}(\dot{\underline{x}}_k)$$

and the measurement deviations

$$\dot{\underline{\delta z}}_k = \underline{z}_k - \dot{\underline{z}}_k.$$

By expanding the vector functions  $f$  and  $h$  in Taylor series and retaining only the first order terms in  $\underline{x}$  one obtains the linearized equations for the Kalman filter.

In the early stages of our research, the system and measurements models were linearized about a nominal trajectory and a linearized Kalman filter was employed to estimate the states. Results were poor so our efforts were directed towards the development of an extended Kalman filter.

#### Extended Kalman Filter

As an alternative to the linearization procedure described above, one may choose to linearize the dynamical and measurement equations about the current state estimates. The resulting continuous-discrete filter [4] is as follows:

Propagation phase

$$\dot{\underline{x}}(t) = \underline{f}(\dot{\underline{x}}(t)) \quad t_k \leq t \leq t_{k+1} \quad (19)$$

$$\dot{P}(t) = F(\dot{\underline{x}}(t)) P(t) + P(t) F^T(\dot{\underline{x}}(t)) + Q(t) \quad t_k \leq t \leq t_{k+1} \quad (20)$$

It is noted that the state estimate is propagated using the nonlinear differential equations describing the dynamics of the system with zero system noise.

Update phase

$$\dot{\underline{x}}_k(+) = \dot{\underline{x}}_k(-) + K_k [z_k - h_k(\dot{\underline{x}}_k(-))] \quad (21)$$

$$P_k(+) = P_k(-) - K_k H_k(\dot{\underline{x}}_k(-)) \quad (22)$$

$$K_k = P_k(-) H_k^T(\dot{\underline{x}}_k(-)) [H_k(\dot{\underline{x}}_k(-)) P_k(-) H_k^T(\dot{\underline{x}}_k(-)) + R_k]^{-1} \quad (23)$$

where

$$F(\dot{\underline{x}}(t)) = \left. \frac{\partial f}{\partial \underline{x}} \right|_{\substack{\underline{x} = \dot{\underline{x}} \\ t = t}} \quad (24)$$

and

$$H_k(\dot{\underline{x}}_k) = \left. \frac{\partial h}{\partial \underline{x}} \right|_{\substack{\underline{x} = \dot{\underline{x}}_k \\ t = t_k}} \quad (25)$$

The various symbols appearing in equations (19) - (25) are defined in Table 2.

TABLE 2  
EXTENDED KALMAN FILTER VARIABLES

Symbol	Definition
$F$	Eqn. (24), (n x n matrix)
$H$	Eqn. (25), (m x n matrix)
$Q$	System noise covariance matrix
$R_k$	Measurement noise covariance matrix at $t = t_k$ , m x m matrix
$K_k$	Filter Gain Matrix at $t = t_k$ , (n x m matrix)
$\dot{\underline{x}}_k(+)$	State vector estimate at $t = t_k$ based on measurements at $t_1, t_2, \dots, t_k$ , (n x 1 vector)
$\dot{\underline{x}}_k(-)$	State vector estimate at $t = t_k$ based on measurements at $t_1, t_2, \dots, t_{k-1}$ , (n x 1 vector)
$P_k(+)$	Estimation error covariance matrix at $t = t_k$ based on measurements at $t_1, t_2, \dots, t_k$ , (n x n matrix)
$P_k(-)$	Estimation error covariance matrix at $t = t_k$ based on measurements at $t_1, t_2, \dots, t_{k-1}$ , (n x n matrix)
$n$	Number of states
$m$	Number of measurements

Extended Kalman Filter Designs

Six extended Kalman filter designs have been developed and studied in the present work. Depending on the assumed operational mode of the seeker, they are divided into two categories. In the first family of filters, the seeker is assumed to be tracking the "white rock" with its optical axis nominally pointed at this target. The available measurements are  $\theta_G$ ,  $\Psi_G$ ,  $\dot{\theta}_p$ ,  $\dot{\theta}_y$ , and  $h_{MSL}$ . The kinematical relations for the rotational motion of the missile are expressed in terms of Euler angles (filter 1A), Euler parameters (filter 1B), and Rodrigues parameters (filter 1C). In the second family of extended Kalman filters the seeker is assumed to be caged or fixed in the missile frame during this midcourse portion of the flight. The available measurements are:  $\alpha$ ,  $\beta$ ,  $h_{MSL}$ ,  $\epsilon_1, \epsilon_2, \epsilon_3$ , and  $\epsilon_4$ . The dynamical equation (3) is not utilized. Instead, the angular rates  $p$ ,  $q$ , and  $r$  are assumed accurately known from the body gyros. Noise exists in the attitude channel through the Euler parameter "measurements" directly. In filter 2A, the angular rates  $p$ ,  $q$ ,  $r$  are prescribed deterministically; in filter 2B  $p$ ,  $q$ ,  $r$  are assumed to be contaminated by noise; while in filter 2C two more states are added in order to include the effects of horizontal wind biases. Euler parameters have been used for the parametrization of the transformation matrix  $T_{CB}$  in all filter designs of the second family.

A typical set of states and measurements for one of the filter designs is given in Table 3; others are given in reference [9].

TABLE 3  
FILTER STATES AND MEASUREMENTS

(1A)(2c)		Number of measurements = 5 (1A), 7 (2C)
STATES		MEASUREMENTS
(1A)	(2c)	(1A)
$x_1 = x$	, $x$	$y_1 = \theta_G$
$x_2 = y$	, $y$	$y_2 = \Psi_G$
$x_3 = z$	, $z$	
$x_4 = u$	, $u$	$y_3 = \dot{\theta}_p$
$x_5 = v$	, $v$	$y_4 = \dot{\theta}_y$
$x_6 = w$	, $w$	$y_5 = h_{MSL}$
$x_7 = p$	, $\epsilon_1$	(2C)
$x_8 = q$	, $\epsilon_2$	$y_1 = \alpha$
$x_9 = r$	, $\epsilon_3$	$y_2 = \beta$
$x_{10} = \phi$	, $\epsilon_4$	$y_3 = h_{MSL}$
$x_{11} = \theta$	, wind <sub>x,BIAS</sub>	$y_4 = \epsilon_1$
$x_{12} = \Psi$	, wind <sub>y,BIAS</sub>	$y_5 = \epsilon_2$
$x_{13} = \text{altimeter bias}$	, $\alpha_{BIAS}$	$y_6 = \epsilon_3$
$x_{14} = \theta_G, BIAS$	, $\beta_{BIAS}$	$y_7 = \epsilon_4$
$x_{15} = \Psi_G, BIAS$	, alt. bias	
$x_{16} = \dot{\theta}_p, BIAS$	, $\epsilon_1, BIAS$	
$x_{17} = \dot{\theta}_y, BIAS$	, $\epsilon_2, BIAS$	
$x_{18} = NA$	, $\epsilon_3, BIAS$	
$x_{19} = NA$	, $\epsilon_4, BIAS$	

Computational Accuracy

Single precision arithmetic (16 bit) was used in obtaining the results presented in this paper. For the nominal flight path assumed and the range of values of  $Q$  and  $R$  matrices considered, numerical difficulties have not been observed. In order to recast the filter algorithm to a numerically more stable form the following equation

was used for the propagation of the estimation error covariance matrix:

$$P_k(+) = [I - K_k H_k] P_k(-) [I - K_k H_k]^T + K_k R_k K_k^T.$$

This is algebraically equivalent to Eqn. (22), but is superior in numerical computations since, to the first order, it is insensitive to errors in the filter gain [5]. However, even with the 16-bit machine, comparison of the results showed no appreciable differences in the state estimates or the diagonal elements of the  $P$  matrix. Equation (22) was used in subsequent runs since its solution requires a smaller number of arithmetic operations.

#### Efficiency Considerations

The efficiency of the estimation algorithm is affected by the filter parameters, numerical integration discretization parameter, update frequency and the actual coding of the algorithm.

The number of filter states is the most important parameter affecting the required run time. Keeping it as low as possible is the central goal of any suboptimal filter design. We have kept the dimension of the state vector as low as possible while maintaining equations for system dynamics which are good representations of actual system characteristics. The "truth model" and the filter equations are nearly identical.

The matrix inversion appearing in the estimation error covariance matrix update equations, Eqn. (23), can be avoided by processing the measurements one at a time [4]. This measurement data processing mode has been implemented in the filter design 1A. Comparison with the standard algorithm, that includes the "exact" inversion of an  $m \times m$  matrix where  $m$  denotes the number of measurements, showed a slight increase in the required run time. An advantage of eliminating the matrix inversion is the reduction of the required computer memory by avoiding the supply of a matrix inversion routine. Since memory considerations are less critical than timing requirements in the present application, no further consideration was given to processing the observations one at a time.

An update period of  $h = 1$  second was found adequate for the fourth order Runge-Kutta numerical integration procedure used during the propagation phase. A smaller step size had a negligible effect on accuracy and increased processing time. The fourth order Runge-Kutta integration procedure, although very accurate, is relatively inefficient for the solution of Eqns. (19) - (20) since it requires the evaluation of the right hand sides four times in each step. A lower order Runge-Kutta method or other integration techniques [7, 8] would require a smaller processing time per step but they would call for a smaller step size.

A significant reduction in processing time is achieved by partitioning the state vector into two parts: one whose elements are time varying and another whose elements are time invariant. The estimation error covariance matrix is partitioned correspondingly. The technique leads to very efficient propagation of state estimates and error covariance matrix as can be seen in Table 4.

Table 4 summarizes some timing data for the various versions of the program for filter design 1A.

TABLE -  
FILTER 1A

Total segment flight time = 70 sec.  
Total segment flight length = 7,000m  
Measurements every 1 second

	Run time reduction. Percentage of the run time required in BASELINE	Required run time on VAX 780 (min.)
Direct coding of filter with extensive use of matrix operation subroutines. (BASELINE)	--	2.70
Partition of the state vector and the estimation error covariance matrix into parts corresponding to varying and time invariant elements during the propagation phase of the Kalman filter algorithm.	31.4%	1.85
Replacement of the matrix operation subroutine by efficient use of in-line loops.	34.1%	1.22

TABLE 4 (continued)

	Run time reduction. Percentage of the run time required in BASELINE	Required run time on VAX 780 (min.)
Optimization of the object program by selecting appropriate compilation options (suppression of line number traceback, etc.).	4.9%	1.16
Decoupling of states during the update phase and assuming $P_{VN} = P_{NV} \approx 0$ and $P_{NN} = P_{NN}(t = 0)$	22.0%	0.905

Estimation Accuracy

The performance of the six extended Kalman filter designs were studied in reference [9] for sensitivities to various levels of system disturbances and initial conditions  $x_0$  and  $P_0$ . The nominal trajectory was as shown in Figure 1. The errors in the state estimates are plotted along with the square root of the variances obtained from the covariances matrix. The estimation error is the difference between the actual state and the state estimate obtained from the filter. All the filter designs were tested using synthetic data and consequently the actual states were readily available for the calculation of the estimation error.

In this study, the missile was assumed to fly about a nominal speed 100 m/s, nominal altitude 200 m, and a nominal pitch angle 5 degrees (Figure 1). Some typical results are now given.

The results shown in Figures 2 - 6 were obtained using filter 2C with the following measurement statistical parameters:

Measurement	Noise standard deviation	Bias standard deviation
$\alpha$	$8 \times 10^{-4}$ rads	0.001 rads
$\beta$	$8 \times 10^{-4}$ rads	0.001 rads
$h_{MSL}$	1.0 m	2.0 m
$\epsilon_1$	$10^{-3}$	0.001
$\epsilon_2$	$10^{-3}$	0.001
$\epsilon_3$	$10^{-3}$	0.001
$\epsilon_4$	$10^{-3}$	0.001

For this test, the total flight time was 90 sec., the total flight length 9,000 m, and measurements were available every 1 sec. To study the effect of errors in the initial state estimates, the values listed in Table 5 were assigned as initial conditions. It is seen that the estimates improve considerably as the "white rock" is approached. Other filter mechanizations performed similarly. The inclusion of noise in the prescribed angular rates in filter 2B did not have a significant effect on the estimator.

TABLE 5  
FILTER 2C, INITIAL CONDITIONS

State	$\dot{x}_0$	$P_0$
$x_1$	-9100. m	$10,000. \text{m}^2$
$x_2$	25. m	$900. \text{m}^2$
$x_3$	-225. m	$900. \text{m}^2$
$x_4$	95. m/s	$25. \text{m}^2/\text{s}^2$
$x_5$	5. m/s	$25. \text{m}^2/\text{s}^2$
$x_6$	5. m/s	$25. \text{m}^2/\text{s}^2$
$x_7$	0.	$5 \times 10^{-4}$

TABLE 5 (continued)

State	$\hat{x}_0$	$P_0$
$x_8$	0.	$5 \times 10^{-4}$
$x_9$	0.	$5 \times 10^{-4}$
$x_{10}$	0.98	$5 \times 10^{-4}$
$x_{11}$	1. N/kg	$1. N^2/kg^2$
$x_{12}$	1. N/kg	$1. N^2/kg^2$
$x_{13}$	0. rad	$5 \times 10^{-5} rad^2$
$x_{14}$	0. rad	$5 \times 10^{-5} rad^2$
$x_{15}$	0. m	$10. m^2$
$x_{16}$	0.	$5 \times 10^{-5}$
$x_{17}$	0.	$5 \times 10^{-5}$
$x_{18}$	0.	$5 \times 10^{-5}$
$x_{19}$	0.	$5 \times 10^{-5}$

#### Sensitivity to System Disturbances

Numerical experiments have been performed with the elements of matrix Q multiplied by 4 and 0.25. No appreciable differences in the estimation errors were observed. Naturally, the uncertainty in the system was increased or decreased respectively during the flight but the estimation errors close to the waypoint were not appreciably influenced.

The actual level of wind disturbances was found to have a significant effect on the filter performance. The estimated wind biases are in excellent agreement (errors well under  $1 m/s^2$ ) with the actual wind biases for winds up to 5 m/s. However, the estimates deteriorated for higher wind velocities. Figures 7-8 show the estimation errors corresponding to wind biases of 10 m/s. It is evident that in this case, the extended Kalman filter is optimistic in the sense that the estimation errors in the y-direction wind bias (state variable  $x_{12}$ ) are consistently larger than that predicted by the error covariance matrix. Short period gusts are not important and wind biases up to 20 m/s with sufficient accuracy for good waypoint navigation.

#### CONCLUSIONS

Any of the six extended Kalman filter designs evaluated in our study can be used for midcourse navigation, heading update, and time-to-go calculations. All designs do not achieve the same navigational performance level. However, positions and attitudes are estimated by all filters with the same accuracy level in the absence of wind biases.

The most significant run time savings are achieved by partitioning the state vector and the estimation error covariance matrix into varying and time invariant elements during the propagation phase of the filter algorithm and by replacing matrix operation subroutines by in-line loops. The results show that the computer with speed of a VAX 780 programmed in FORTRAN should be able to perform all calculations fast enough for this algorithm to work in real time.

The assumption that the angular rates p, q, and r are known (filter 2C) reduces significantly the dimension of the estimators and hence, the CPU requirements. Since fairly accurate measurements of p, q, r were assumed available from body gyros, the estimator 2C with only attitude measurement noise gave the same accuracy as those retaining the full rotational dynamical equation (filter 1A). If the orientation parameters are accurately computed based on prescribed p, q, and r, the rotational kinematical equations may also be eliminated.

Among the six estimators studied, filter 2C has the following advantages: (1) give good estimates of wind biases; (2) utilizes Euler parameters for the parametrization of the rotation matrix, thus reduces the computational burden; (3) avoids the singularities which are present in the Euler angles or Rodrigues parameter formulations. The filter is insensitive to inaccurate descriptions of the system and measurement noise but its performance begins to deteriorate when the wind biases are in the neighborhood of 10 m/s. This deterioration is not significant in the application at hand.

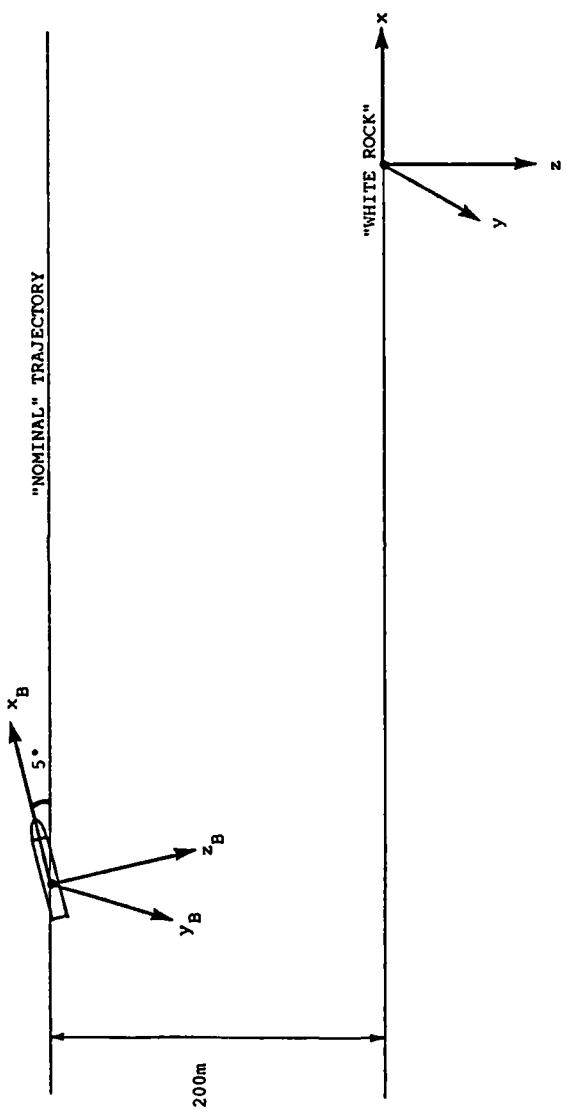


FIGURE 1. Missile "Nominal" Trajectory

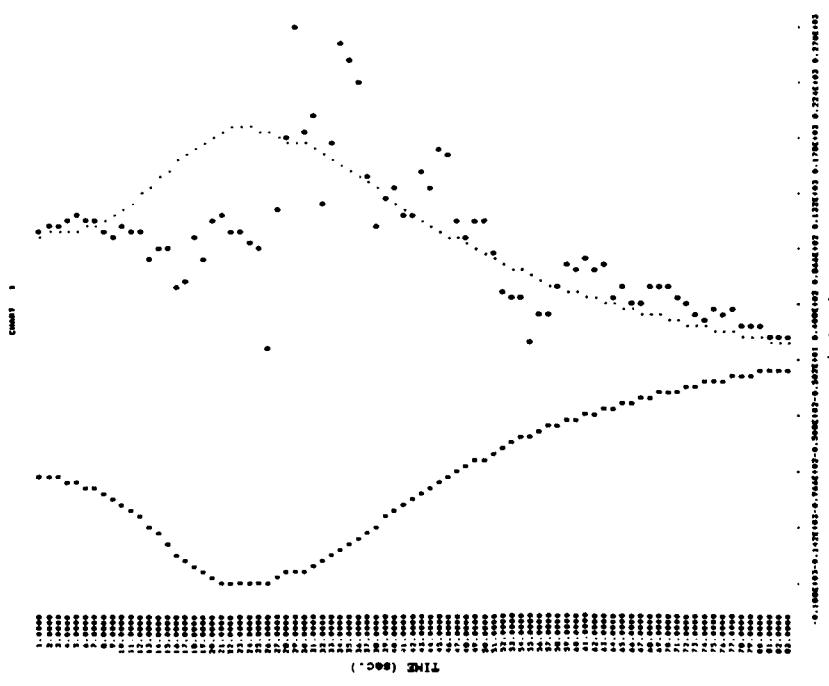


FIGURE 2. FILTER 2C, STATE 1

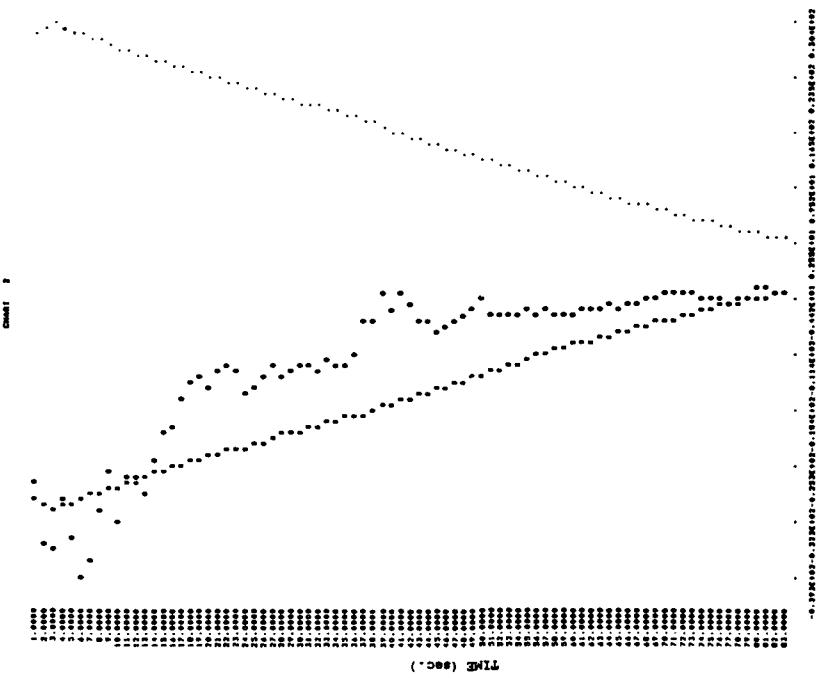


FIGURE 3. FILTER 2C, STATE 2

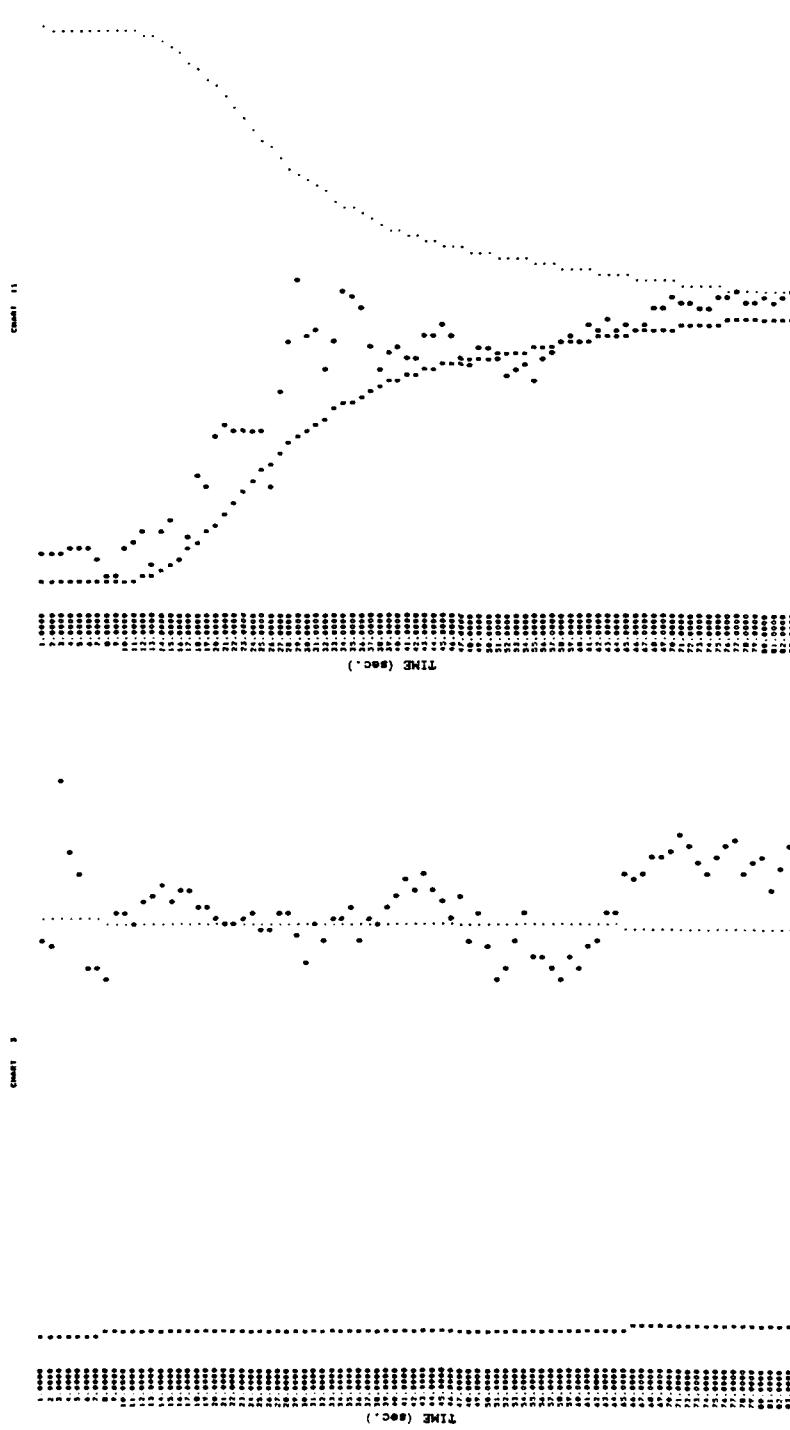


FIGURE 4. Filter 2C, State 3  
2 (meters)

FIGURE 5. Filter 2C, State 11  
X-TWANG LUNG (M. Hg.)

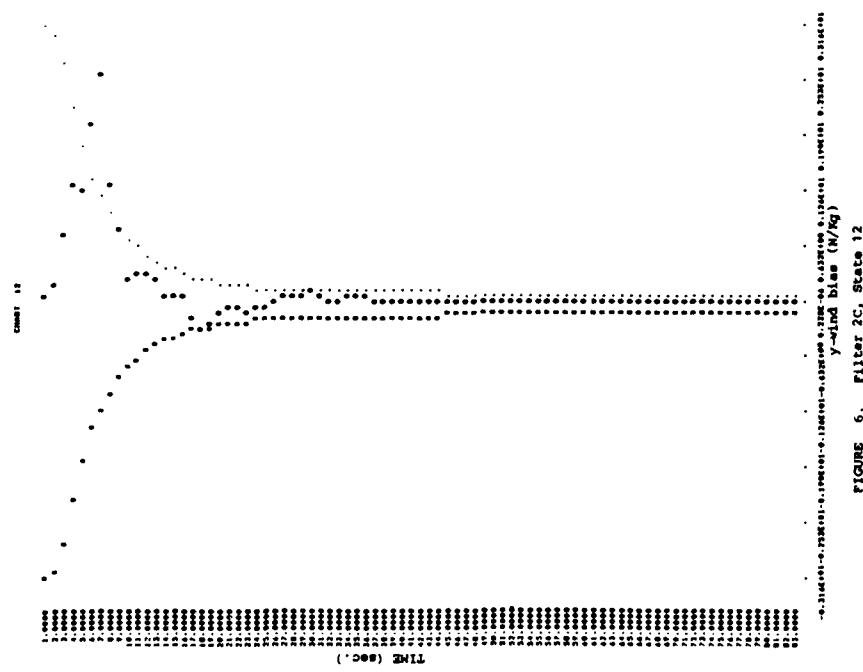


FIGURE 6. Filter 2C, State 12

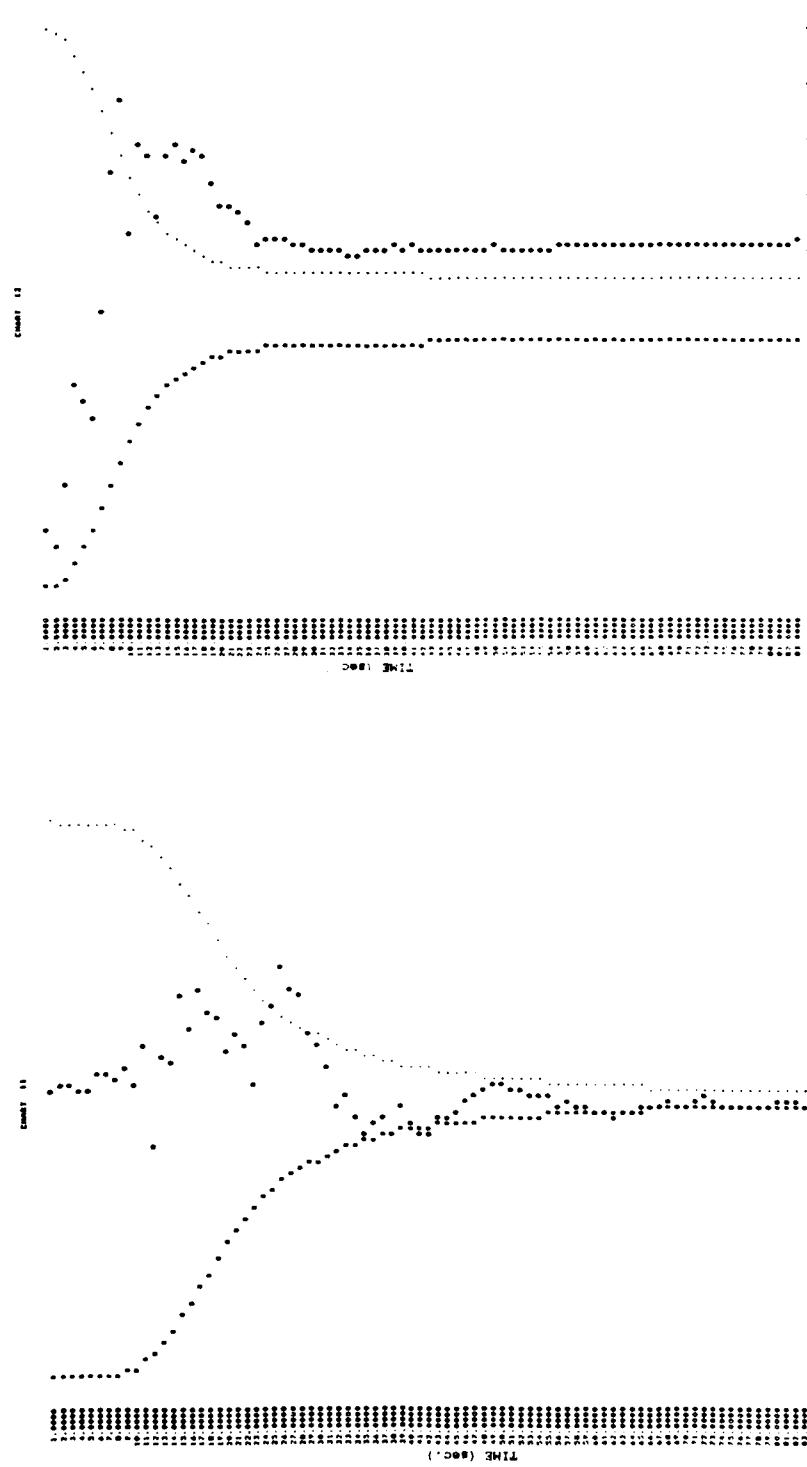


FIGURE 7. Filter 2C, State 12 (high wind disturbances)

FIGURE 8. Filter 2C, State 12 (high wind disturbances)

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TACTICAL MISSION PLANNING AND MANAGEMENT  
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SUMMARY

This paper summarizes the operational requirements for automation of tactical mission planning and management. Based on these requirements the CAMPAL-system is reviewed which has presently been realized by the NLR in co-operation with the RNLAF. Some analysis software packages are highlighted since they provide the CAMPAL-system with unique planning features for penetration and attack. Finally attention is paid to the promises for tactical mission planning and management as suggested by recent advances in the area of knowledge engineering and Expert Systems.

1. INTRODUCTION

The tactical mission scenarios for the 1990's time frame can in general be characterized by the presence of an intense ground-to-air and air-to-air threat and the requirement to operate at night and in adverse weather conditions. Attacks have to be carried out on highly defended ground-targets through a hostile network of integrated defences.

In providing the next generation aircraft with the capabilities to operate in this complex environment, the emphasis is mostly laid on the aircraft side by modifying existing aircraft designs or developing new aircraft concepts. Also new stand-off and unmanned weapon systems are developed for particular missions.

In our opinion not enough attention has been paid to another aspect, more and more vital to the overall mission performance: tactical mission planning and management, and the automation of these processes.

The dynamics of the battlefield and the increased enemy defence capability have led to the conclusion that, without the aid of an automated planning system, penetration and attack analysis are highly inefficient. Automation of the planning process must give the pilot/planner all opportunities to concentrate, within the limited time available, on the essentials of his mission plan. On top of that, without an automated planning system it has become increasingly difficult to achieve the accuracy and completeness required for a cost-effective use of the new avionic potential of modern aircraft/weapon systems.

2. REQUIREMENTS FOR AUTOMATED MISSION PLANNING AND MANAGEMENT

The operational procedures in preparing tactical missions have been the subject of several studies [1, 2]. From these studies the following key requirements are derived for automated systems for mission planning (these requirements are equally valid for the similar process of tactical in-flight management):

- quick provision of planning information,
- display capability of planning information,
- interactive planning capability,
- on-line mission analysis capability,
- quick generation and transfer of planning results,
- mission preview and review capability.

2.1 Quick provision of planning information

It is evident, although mostly not realized, that the planning of tactical missions should incorporate all available information and that this information needs to be reliable and up-to-date. There are three general categories of planning information:

- o Mission related data, which are needed to activate the planning process and become available after acceptance of the air task:
  - tasking information  
(target, alternate targets, times, communication and control data),
  - force allocation  
(number of type of aircraft, pilots, a/c configurations),
  - target information  
(collected and prepared by intelligence officer).
- o Situation related data for the area of mission interest. This information has a medium to very high update rate and needs to be updated even during the mission planning process:
  - intelligence information  
(fixed and tactical SAM-sites, EW-sites, GCI and enemy air defence bases, front line own troops),
  - navigational restrictions  
(prohibited areas, safe lanes),
  - meteorological conditions  
(status of own and alternate airbases, expected enroute and target weather).
- o Permanent planning data with a low update rate:
  - aircraft performance data,
  - aircraft standard configuration data,
  - navigational aids,
  - standard check- and waypoints,
  - friendly airbase data,

- navigational obstructions and obstacles,
- navigation maps,
- digitized terrain data (elevation and features),
- defence system characteristics,
- tactical advisory guidelines,
- weapon effectiveness data.

Assistance to the planner in gathering and interpreting this information is given by intelligence, navigation and meteo specialists. It is expected that these officers will be supported in the near future, similarly as the mission planner, by automated systems. For quick provision of up-to-date information there is a need to create a network at airbase level which links the mission planning system to other systems. The permanent databases can be provided either by systems resident databases or by databases shared by the subsystems within this network.

Since most of the mission planning databases are classified, special attention should be given to the protection of this information.

## 2.2 Display capability of planning information

The planning information has to be displayed in a surveyable manner in order to aid the planner in data interpretation, interactive routebuilding, attack planning, etc. It is obvious that for visual routing geographical, map related information plays the most important role. There are two types of geographical information available for computerized use:

- o Structured geographical information.  
In a database with structured geographical information, each object (e.g. roads, bridges) is stored together with its relevant attributes (e.g. name, co-ordinates and size). Structured geographical information is characterized by a great flexibility in use and presentation. However, composing and updating a database with this type of information is very labour-intensive.
- o Non-structured geographical information.  
This type of information can only be used in mission planning for the projection of aeronautical maps as background on a graphical display (digital map projector). Digitizing (scanning) the aeronautical maps is less labour-intensive but requires a large storage capacity. For storage of scanned map information use can be made of "digital optical recording" techniques. Advantages of this technique are its high reliability and low storage costs per bit.

Presentation of geographical information must be done on large graphic colour displays. Separate overlays can be used to display graphically the information of the planning databases as selected by the planner (e.g. meteorological data, threat situation, terrain contour and masking diagrams). In addition an alphanumeric display can be used to display associated mission parameters. The graphic display must provide an overview of the area of mission interest and must enable a co-ordinate pointing accuracy at least better than the waypoint accuracy of the aircraft/weapon navigation system. High resolution with easy zoom and move capability are needed to meet these requirements.

## 2.3 Interactive planning capability

It is essential for mission planning that mission definition stays under control by the planner/pilot and that the judgement, experience and intuition of other specialists involved in mission planning can be used directly in the planning process. From the user acceptance point of view the need is stressed for use of the system by mission planners who may not have a sophisticated knowledge of computers. For that purpose special attention has to be paid to user-friendly interfaces in the interactive dialogue: command languages, menu's, function key-boards etc. The provision of user-friendly interfaces is essential in case of the use of Expert Systems which makes available expert knowledge to a relatively unskilled planner or to planners under stress and in state of fatigue.

## 2.4 On-line mission analysis capability

Mission planning can in the first place be speeded-up by supporting the route building process with an automatic calculation of:

- aircraft performance data  
(range limitations, bingo and joker fuels, take-off and landing limitations),
- navigation plan data  
(headings, times, distances, safe area and minimum enroute altitudes),
- co-ordinate transformations  
(Lat/Long-UTM conversion, range/bearing calculations, composition of the flightplan).

In addition to just speeding-up the planning, significant improvement can be realized by interactive terrain analysis and defence threat assessment. Three modes of terrain and threat analysis can be distinguished: the display, the evaluation and the advisory mode. Especially in the latter two modes the application of Expert Systems can support the planner's decision making process by presenting validated information in a timely manner and by making rapid assessment of alternative routes and tactics:

- c Display mode.  
In the display mode the mission planner selects the types of defence threats to be shown and their way of presentation on the graphic display: terrain masking diagrams, threat coverage diagrams or radar coverage diagrams. Route selection is done, totally based on the information displayed on the graphic display. Calculation of the masking and coverage diagrams have to incorporate the latest known intelligence information (e.g. defence system position and operational status, defence system characteristics), aircraft flight profile, terrain elevation and terrain features.

- o Evaluation mode.

After the planner has generated route and tactics, the planning system will summarize in the evaluation mode the known enemy threats along the intended route. Furthermore, in the evaluation mode the system provides the planner with relative indications on threat exposure times and expected aircraft safe crossing probability. This allows the planner to generate better route alternates.

- o Advisory mode.

Terrain analysis combined with defence threat analysis can automatically generate low altitude flight trajectories and tactics with minimum exposure to both fixed and mobile threats. This mode will enable quick planning since at most minor adjustments are needed before acceptance by the planner. The advisory mode for route building may also be of great value in preparing low-level missions for the next generation aircraft, equipped with integrated navigation and terrain following systems. Limitations in on-board data storage capacity may require the preparation of missionized data loading (e.g. GPS and DLMS data).

To speed-up and improve the planning process further it is also needed to perform analysis functions during detailed planning of attack and weapon delivery.

Automated weapon effectiveness considerations, using the experience of weapon instructors and senior pilots, must give the planning crew advisory guidance for optimal target, weapon and delivery matching. This guidance must include aspects as target acquisition, the use of special run-in tactics and the coordination within large attack forces.

#### 2.5 Quick generation and transfer of planning results

Traditionally much effort in mission planning has to be spent on the preparation of the pilot knee-board checklist and navigation maps. The checklists are used for the set-up and operation of the aircraft on-board systems. On the maps all the mission relevant data are drawn as collected or calculated during the planning.

Automated mission planning should at least provide a hard-copy of checklists and colour maps. The same formats and symbology should be used as for the corresponding cockpit interfaces.

Preflight set-up of the aircraft systems is often still performed by manually typing in the data via cockpit keyboards. This requires too much time, causes pilot fatigue and is very susceptible to errors. With the advent of computer intensive aircraft, it is highly desirable to insert the mission planning data automatically into the on-board systems.

There are several methods for doing this:

- Data Transfer Module (DTM)  
(ground-crew or pilot transferred, non-flight or flight worthy DTM),
- Radiating RF link,
- Fixed RF cable.

Mechanization of the data transfer by a pilot transported flight worthy DTM is probably the most advantageous solution. Loading of the DTM could be accomplished at the mission planning station or in the pilot briefing room and the loaded DTM could be given directly to the pilot. The DTM would be carried to the aircraft by the pilot and inserted into a convenient cockpit receptacle.

In addition to the transfer of mission planning a DTM may relieve limitations in on-board data storage by selective loading of missionised computer programs and databases. Also the possibilities of in-flight retasking, in-flight recording and post-flight data recovery (tactical and maintenance data) are enhanced by a DTM.

#### 2.6 Mission preview and review capability

Once the mission data are established in a groundbased mission planning station, a next logical step is to "prefly" planned missions. Pilots could use simulations (on the mission planning station or on on-board displays) to familiarize themselves, faster than real-time if desired, with critical parts of the mission. By preflying their mission successfully, pilots could increase their confidence and effectiveness.

For in-flight operation modern fighter cockpit concepts (hands on controls, especially in single pilot operation) prohibit the use of hand held maps or other loose paper work prepared during pre-mission planning. The need is evident for appropriate on-board (map) displays presenting the navigational and tactical information which keeps the pilot informed about the mission progress or allows him to deviate adequately from the original mission plan.

Future mission scenarios tend to be very changeable so pre-mission planning data should be automatically updated during mission execution with real-time data from external sources (such as AWACS). Quick evaluation and presentation of relevant information on on-board real-time displays must enable a flexible in-flight response of the aircrew to fast developing new tactical situations.

### 3. COMPUTER AIDED MISSION PREPARATION AT AIRBASE LEVEL (CAMPAL)

To illustrate a concept for automated mission planning the CAMPAL-system will be briefly described here. As far as we know this is the only existing system which meets the requirements as given in chapter 2.

The CAMPAL-system is the result of an incremental development by the Royal Netherlands Air Force (RNLAF) and the National Aerospace Laboratory (NLR). The RNLAF contribution is based upon its know-how of the actual ground operations and air operations. The NLR contribution is a logical extension of its knowledge on the military organization, and its experience with operational evaluation, integrated combat training, and automation.

At the moment this system is operational for RNLAF use in both war- and peace-time mission preparation. Its essentials are:

- it speeds up preparation by automation of standard procedures,
- it improves preparation through easy availability of the most recent information, based on specific weapon-systems characteristics and the ever changing scenarios,

- it facilitates preparation because the pilot can interactively consult (no computer knowledge required) various relevant data sources (intelligence, standing procedures, meteo, geographical) before taking his decisions,
- it strengthens the preparation by automatically assessing the threat figures using the complex, quickly changing intelligence scenario in full detail,
- it is tuned to the specific F-16 demands for operation in the European environment.

The current CAMPAL operational hardware is based on a VAX 11/780 computer and an interactive graphic workstation Sigmex 7000 (Fig. 1). At least three workstations can be served by one computer. The large bulk of geographical data is stored on the Philips Digital Optical Recorder (DOR). The CAMPAL-system is part of the RNLAF Airbase Command, Control and Information System (ABCCIS).

The planning process is controlled via the alpha-numerical terminal of the interactive workstation (Fig. 2). For the actual route determination, the workstation includes a tracker ball. The digitizer serves as an additional means of input. On a raster-scan colour monitor, the map information and related data are displayed. These data can be processed via the colour hardcopy unit. The graphic printer produces all planning results other than the navigation maps, such as the F-16 navigation system (INS) data, the F-16 weapon-system (SMS) data, the alpha-numerical navigation information for the mission folder, and general mission data. In fact CAMPAL produces fully automatically the Combat Mission Folder (CMF), which is standardized for F-16 operations in Europe (Fig. 3).

#### 4. MISSION ANALYSIS MODELS

The development of the CAMPAL-system has become possible through the results of years of in-depth research. Some topics of this research resulted in the following analysis models:

- Enemy Defence Analysis Models (EDAM),
- Digital Terrain Analysis Models (DITAM),
- Munition and Delivery Analysis Models (MADAM).

Since these models provide the CAMPAL-system with some unique analysis features for the penetration and attack planning, they will be highlighted in this chapter.

##### 4.1 Enemy Defence Analysis Models (EDAM)

The EDAM software package has been developed in order to assess the defence system capabilities and to evaluate tactics for low-altitude missions in an environment of ground-based air-defence systems. The basic problem solved by EDAM is to find the best compromise between the various, often conflicting mission parameters, such as penetration speed and altitude, type of formation, crossing distance to the defence system, attack profile and heading in order to assure maximum aircraft survivability and mission effectiveness.

EDAM basically deals with three sets of parameters each describing an essential component in the aircraft/defence system interaction:

- the defence system data (e.g. detection performance, fire control computer modes, IR capabilities, missile kinematics and missile guidance dynamics),
- the environmental factors (e.g. terrain features, meteorological conditions),
- the aircraft characteristics (e.g. speed and altitude profile, IR signature, radar cross section, ECM capabilities).

There are many ways to present the EDAM results. An often chosen approach is to present aircraft survivability curves as a function of crossing distance to the defence system for each set of parameters (Fig. 4). To be usable as input for CAMPAL-algorithms these data are converted into (Fig. 5):

- a minimum and maximum effective range of the enemy defence system,
- a risk figure accounting for defence system effectiveness for the exposure time of the aircraft within the effective ranges of the system,
- tactical recommendations which can be translated into advisory guidelines on tactics to the mission planner.

The tactical recommendations provide the most authoritative guidance available. Beside on the EDAM results, they are based on the expertise of senior pilots, weapon and tactical officers. They are also incorporated in daily training and are kept up-to-date with the results of trials and the latest intelligence information.

EDAM dynamic programming algorithms generate minimum risk routes through enemy defences for a given scenario and enable quick planning by providing automatically best routes and alternates to the mission planner. The probability of safe crossing for a given threat and route is calculated according to the expressions in Fig. 6.

##### 4.2 Digital Terrain Analysis Models (DITAM)

DITAM is a software package covering all applications in the mission planning domain utilizing the Digital Landmass System (DLMS) database. In comparison to other often specialized digital terrain databases the DMA produced worldwide DLMS database is probably the most widespread database and the one being considered for use in most near and long term applications.

The DLMS database contains two parts: the Digital Terrain Elevation Data (DTED) and the Digital Feature Analysis Data (DFAD) both in two levels of resolution. For example DTED level I and II correspond to a resolution in the horizontal plane of approximately 100 and 30 meters respectively [3].

The DITAM package uses both parts of the DLMS database to provide the mission planning process with:

- o Coverage diagrams for defence systems with known co-ordinates and sensor capabilities. Low altitude missions can take great advantage during penetration and attack of the limited coverage of the defence system due to terrain blocking the geometric line-of-sight or the electro-magnetic path between the system associated radar and its target. Parameters in the masking diagrams provided by DITAM are (Fig. 7): type of sensor (visual, TV, IR or radar), system co-ordinates and antenna height, meteorological conditions and target flight altitude.

Studies and validation with real radar scope images have confirmed that for radar detection not only terrain masking but also ground-clutter is a determining site specific phenomenon. Knowing the radar parameters and the site features the DITAM radar model predicts low grazing angle ground-clutter (Fig. 8).

Once the coverage by the systems affecting the planned route is known, the EDAM calculations can be updated with a more accurate estimate of the threat exposure times.

- o Threat "density" diagrams for an area with mobile enemy defence systems. This DITAM feature is important e.g. for the planning of the penetration through the forward edge of battle area where a high density mobile threat, allocated to the front divisions, is expected. These threat density diagrams (Fig. 9) take into account the latest known system position, the mobility of these systems, the terrain masking and the terrain trafficability.
- o Visual map presentation, an option of DITAM still under development. The DLMS database has enough resolution and detail for a pictorial presentation of the real terrain. To this purpose the DFAD level I second edition will include in the database the lines of communication: i.e. roads, railroads, rivers, etc. Visual digital maps with perspective views permit the mission planner to examine in detail the route, waypoints, off-set aim points and to preview the target area and other critical parts of the mission.
- o Safe altitude figures for low level navigation. DITAM calculates for each route leg on a low level mission the safe altitude figures like Safe Area Altitude (SAA) and Minimum Enroute Altitude (MEA). For example the SAA is a standard figure and has to be determined by taking the highest terrain or obstacle within 10 nm of the intended route, adding 500 ft and then rounding off to the next 100 ft level. To this purpose DITAM combines the DLMS terrain data with a separate file containing the latest known vertical obstacle data such as man build features, radio towers, chimneys, power lines etc.

#### 4.3 Munition And Delivery Analysis Models (MADAM)

The MADAM package is an interactive weaponeering computer program. It is used in a stand-alone version for weaponeers and targeteers and in a dedicated version for mission planners (CAMPAL). Using MADAM, weapon effectiveness can be calculated, force requirements can be determined and attacks can be optimized. To that purpose MADAM comprises the following elements:

- a weapon effectiveness module based on the standard Joint Munitions Effectiveness Manual (JMEM) methodologies [4],
- a weapon trajectory calculation module, for F-16 planning based on the F-16 Fire Control Computer (FCC) algorithms [5]
- a target datafile with physical and vulnerability data for various damage criteria,
- a weapon datafile with fusing, fragmentation and *safe escape* data,
- a standard aircraft-weapon configuration datafile,
- a standard attack profile datafile.

The MADAM package has been evaluated extensively at various RNLAf airbases and has proved to be a worthwhile aid for wing weaponeers and squadron pilots. In the CAMPAL version for mission planning an attack can be optimized in a few minutes including determination of the optimal stick length and the generation of sight settings for manual aiming.

For F-16 mission planning, MADAM provides in addition the load inventory and attack profiles for the Stores Management System (SMS).

#### 5. EXPERT SYSTEMS FOR MISSION PLANNING AND MANAGEMENT

Present automation of mission planning is focussed mainly on two aspects:

- speeding-up the planning process by automation of the time consuming standard procedures,
- improving the planning process by adding interactive analysis capabilities for penetration and attack.

The expert knowledge to support these functions is made available as far as this knowledge can be comprised traditionally in a database or can be formulated in mathematical algorithms and stochastical formalisms. The personal assistance of these experts at every planning is still of the uttermost importance. The limited number of specialists available makes this assistance in practice impossible.

Recent advances in the area of knowledge engineering and especially Expert Systems seem to offer a potential solution to alleviate this problem. Expert Systems are software environments designed to aid in solving problems which require a high level of expertise, some degree of inference ("reasoning") and the use of heuristics (non-rigorous procedures or "rules of thumb"). Especially in the multi-disciplinary environment of tactical mission planning, where specialist domain knowledge is essential and judgement, experience and intuition play a larger role than mathematical algorithms and formalisms, they may offer powerful decision aids to the planner.

To indicate the potential of Expert Systems for mission planning and management, the following list gives some examples of application candidates (this list is intended to be representative rather than complete):

- target prioritization,
- air task allocation,
- force allocation,
- check on available resources,
- weather assessment,
- threat lethality assessment,
- penetration route building,
- target-weapon matching,
- optimum attack profile selection,
- advise on tactics and countermeasures,
- evaluation of user-chosen options.

There are presently already in existence or development a number of systems which claim an expert-knowledge performance in the mission planning and management domain:

- a Route Planning Aid for maritime operations [6],
- the Knowledge Based System (KNOBS), supporting at TACC/ATO level the planning of counter-air and interdiction missions by aiding in the selection of targets, aircraft, weapons and the use of ECM [7].
- a Route Planning Aid for the enroute and target area, providing minimum lethality routes based on threat exposure and terrain masking [8],
- The Tactical Air Targeting Recommender (TATR) for the planning at TACC level, aiding in selection and prioritizing airfields and target elements on those airfields [9],
- the DARPA Pilot's Associate Program, an application of expert system technology to several flight domain tasks [10],
- the Knowledge based Weaponning and Targetting at Airbase Level (KWETAL) system, a feasibility study presently being carried out by the NLR; this system supports the air task processing at wing level and the target-weapon matching at squadron level [11].

In principle three types of Expert Systems (all with explanation facilities) can be distinguished:

- decision aids which give the planner at various stages of the planning a list or range of possibilities, if wanted ordered by preference,
- critiquing systems which analyze and criticize the planner's mission plan [12],
- autoplanning aids which tell the mission planner what to do.

In the environment of tactical mission planning we prefer an "intelligent colleague" (= a combination of a decision aid and critiquing system) rather than an autoplanner. Some reasons for this preference are:

- an "intelligent colleague" is less threatening to the planner since the computer becomes an ally rather than a potential competitor,
- there are frequently special circumstances which cause a planner to lean one way or another in a mission plan; many of these are difficult, if not impossible, to anticipate and quantify; as a result, it makes sense to let the planner focus initially on the approach he feels is most appropriate,
- an "intelligent colleague" forces the planner to grapple with the problem himself before turning to the computer for assistance; he must think through the mission and assess any special circumstances which might be present,
- different planners may often approach a particular mission differently; no single approach is necessarily "right" and even approaches which are suboptimal may be adequate,
- the creation of a complete set of rules as needed by an autoplanner to generate optimal solutions would take many years, if possible at all,
- one of the basic rules for mission planning is to be "unpredictable"; an autoplanner generates a predictable and reproducible mission plan,
- the planner/pilot has ultimate responsibility for mission execution; therefore it makes good sense to leave all primary and final decision making with him.

## 6. CONCLUSIONS

- (1) In order to meet readiness criteria for tactical missions in even the most complex intelligence scenarios and operational conditions, more and more emphasis has to be laid on mission planning. Research has shown that requirements for mission planning mainly deal with quick provision and display of information, threat assessment and analysis, and transfer of mission data into the aircraft/weapon system.
- (2) The automated mission planning system which meets these requirements is the CAMPAL-system. It speeds up the planning process by automation of standard operating procedures and improves it by adding interactive analysis capabilities for penetration and attack. For the latter purpose special software packages have been integrated in the CAMPAL-system with unique features for enemy defence analysis (EDAM), for digital terrain analysis (DITAM) and for munition and delivery analysis (MADAM).
- (3) With traditional automation techniques, the personal assistance of various specialists in every mission planning is still needed but can hardly be realized in practice. Recent advances in the area of Expert Systems seem to have the potential to alleviate this problem. In our opinion the main goal of an Expert System should be an "intelligent colleague" rather than a replacement of the mission planner.

## 7. POINTS OF CONTACT

Those who like to have more information on the systems and models described, are advised to contact one of the following points:

- Royal Netherlands Air Force (RNLAF),  
Assistant Chief of Staff for Operational Requirements (AOB),  
Section Operations Research and Evaluation (ORE),  
P.O. Box 20703, 2500 ES The Hague, The Netherlands.
- National Aerospace Laboratory (NLR),  
Flight Division (V),  
Military Operations Research Group (MOR),  
P.O. Box 90502, 1006 BM Amsterdam  
The Netherlands.

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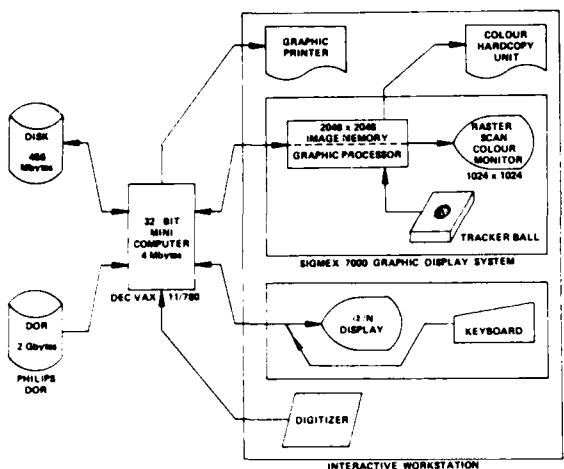


Fig. 1 CAMPAL operational hardware configuration



Fig. 2 CAMPAL interactive graphic workstation

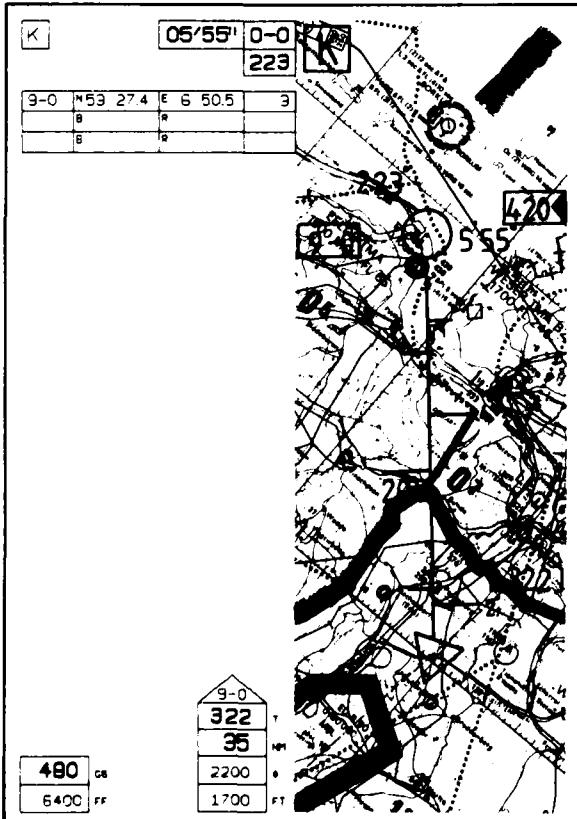


Fig. 3 CAMPAL produced F-16 Combat Mission Folder

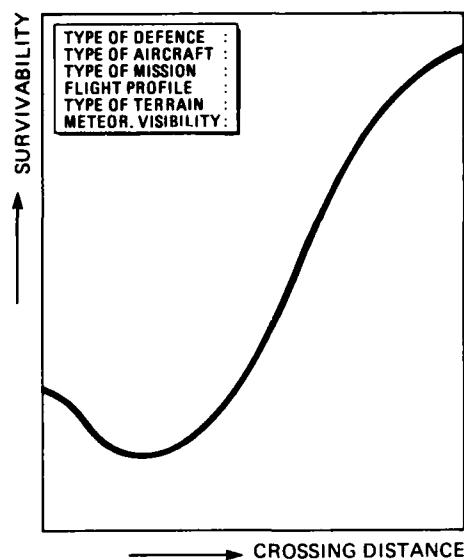


Fig. 4 EDAM aircraft survivability curves

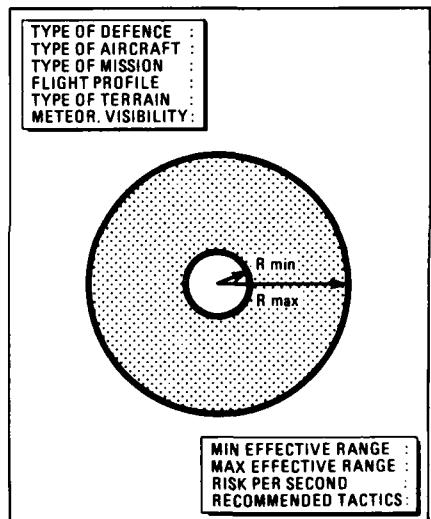


Fig. 5 EDAM data for CAMPAL algorithms

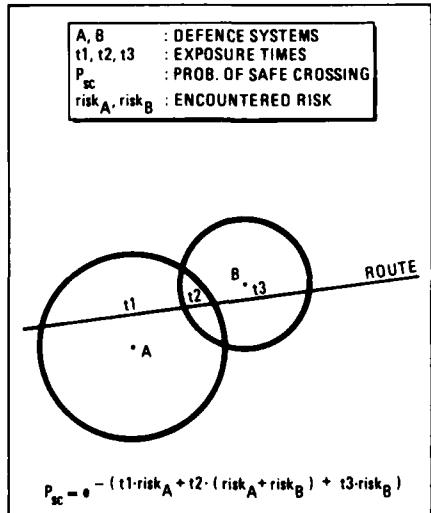


Fig. 6 EDAM calculation of probability of safe crossing

SYSTEM POSITION :  
SYSTEM ELEVATION :  
TYPE OF SENSOR :  
ANTENNA HEIGHT :

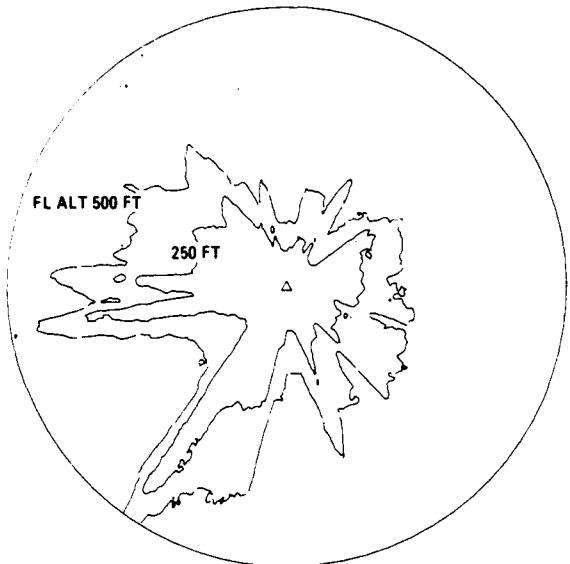


Fig. 7 DITAM coverage diagram (without clutter)

LOW ..... MEDIUM ..... HIGH  
· · · · · · · ·



Fig. 9 DITAM threat density diagram

SYSTEM POSITION :  
SYSTEM ELEVATION :  
TYPE OF SENSOR :  
ANTENNA HEIGHT :



Fig. 8 DITAM coverage diagram (with clutter)

PROBABILITE DE COLLISION DE MISSILES  
DE CROISIERE VOLANT EN FORMATION

Michel FERRAND et Philippe RIOT, LRBA, 27207 VERNON - FRANCE

RESUME :

La loi de collision de missiles de croisière qui volent en formation à la même hauteur de consigne est établie mathématiquement. Le taux moyen de perte pour une mission de plusieurs missiles sur une trajectoire comprenant plusieurs tronçons est ensuite établi (à l'issue de chacun des tronçons, un recalage par corrélation de profils de terrain est supposé effectué). La présentation se termine par une application numérique.

1 - INTRODUCTION

La mission typique d'un missile de croisière consiste à suivre à très basse altitude une trajectoire préprogrammée l'amenant de son point de lancement à sa cible. Le guidage dans le plan vertical utilise essentiellement un radar de suivi de terrain, et dans le plan horizontal une centrale inertie, laquelle conduit malheureusement à des erreurs de localisation croissant avec le temps. Il y a donc généralement nécessité de recalages périodiques de la navigation, recalages que l'on peut effectuer par corrélation de profils de terrain dès lors que l'on dispose de cartes de relief des zones de recalage et que l'on mesure le relief du terrain survolé à partir d'un radioaltimètre et de la chaîne baro-inertie de la centrale de guidage.

L'imprécision au but due à la dérive inertie entre instant de dernier recalage et instant d'arrivée sur la cible, d'une part, les risques de destruction du missile inhérents au suivi de terrain, d'autre part, font qu'il peut être utile de tirer plusieurs vecteurs sur une même cible pour augmenter le taux de succès de la mission.

Par ailleurs, l'attaque en salve présente un avantage supplémentaire : elle permet de mieux pénétrer les lignes de défense adverses en saturant les moyens d'identification et de poursuite. Cet effet de saturation sera d'autant mieux assuré que les missiles seront spatialement peu dispersés.

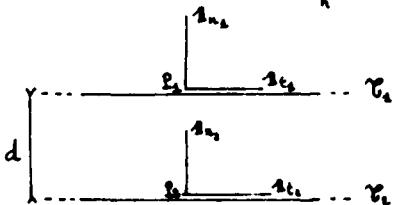
Les vols en formation rapprochée de missiles de croisière apparaissent donc intéressants au plan tactique. Mais les imprécisions de navigation des missiles font que malheureusement les risques de collision ne peuvent être négligés. Le but de l'étude présentée est donc l'analyse de ces problèmes de collision entre missiles de croisière volant en formation. Le plan adopté est le suivant :

- Chapitre 2 : collision de deux missiles dans le plan horizontal
- Chapitre 3 : collisions de n missiles sur l'intervalle (0, t)
- Chapitre 4 : cas général. Salve de n missiles, trajectoire en k tronçons
- Chapitre 5 : application numérique
- Chapitre 6 : conclusion

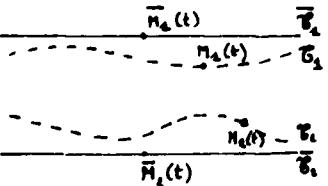
2 - COLLISION DE DEUX MISSILES DANS LE PLAN HORIZONTAL

Définissons géométriquement la notion de vol parallèle. Soit  $\bar{T}_1$  la trajectoire nominale du premier missile ; à chaque point P de  $\bar{T}_1$  est associé le repère canonique  $R_1 = (P_1; \bar{1}_{t1}, \bar{1}_{n1})$ .

Le second missile suit une trajectoire  $\bar{T}_2$  telle que son repère canonique  $R_2$  à l'instant t se déduit de celui de  $\bar{T}_1$  au même instant par la translation  $U = -d\bar{1}_n$  par rapport à  $R_1$ .



La configuration lors d'un vol réel se présente de la manière suivante :



Les trajectoires et les positions nominales sont notées  $\bar{T}_1, \bar{M}_1(t), \bar{T}_2$  et  $\bar{M}_2(t)$  et leurs équivalents effectifs  $T_1, M_1(t), T_2$  et  $M_2(t)$

Posons :

$$\bar{M}_1 M_1(t) = E_1(t)$$

$$\bar{M}_2 M_2(t) = E_2(t)$$

ce sont les écarts de route.

Rappelons que

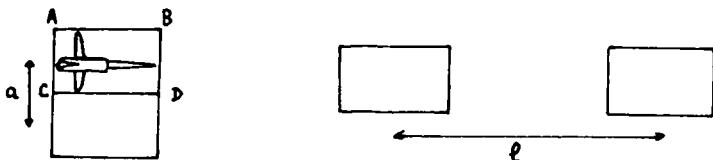
D'après la relation de Chasles :

$$E_2(t) = E_1(t) + M_1 M_2(t) - d \bar{1}_n$$

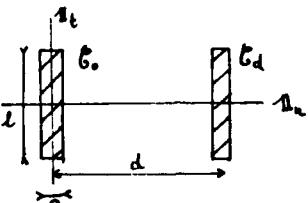
Il y a collision dans le plan horizontal si  $M, M_2(t) \in \mathcal{C}_0$  domaine que nous allons définir. Appelons  $\mathcal{C}_d$  le translaté ( $\mathcal{C}_0 + d \mathbf{I}_n$ ). La collision se produit si  $E_1(t) - E_2(t) \in \mathcal{C}_d$   
Remarque : les missiles volent à la même hauteur au dessus du terrain, cette hauteur devant être la plus faible possible, de façon à limiter au maximum la détectabilité des missiles, mais devant également être compatible avec les exigences de suivi de terrain.

Revenons à la définition du domaine .

L'expansion dans le plan horizontal de la cellule du missile est circonscrite par un domaine rectangulaire ABCD. Nous admettons qu'il y a collision si les domaines rectangulaires respectifs des deux missiles entrent en contact selon les côtés AB (ou CD) ou bien si les côtés AC (et BD) sont parallèles entre eux et distants d'au plus 1.



Cette configuration-ci se justifie par la présence des jets de tuyère qui exclut tout contact en "file indienne". Le domaine de collisions  $\mathcal{C}_0$  est ainsi caractérisé.



Nous supposons que  $l \gg a$  de telle sorte que  $\mathcal{C}_0$  et  $\mathcal{C}_d$  sont assimilés aux segments de droite qui subsistent lorsque  $a$  tend vers zéro.

La probabilité de collision sera déterminée si l'on connaît la fréquence de passage par  $\mathcal{C}_d$  de la trajectoire du processus  $\mathcal{E}(t) = E_1(t) - E_2(t)$  dans un intervalle de temps  $[t_1, t_2]$ .

Cette probabilité de collision  $P(t)$  sur l'intervalle  $[0, t]$  est égale au nombre moyen de passage de  $\mathcal{E}$  par  $\mathcal{C}_d$ .

Nous admettons que l'écart  $E_1(t)$  est dû pour l'essentiel à la dérive des deux inerties, donc  $\mathcal{E}(t)$  est égale à la variation relative des erreurs inertielles de position entre deux recalages consécutifs dans le repère tangent à la trajectoire de l'un des deux missiles. Comme l'on raisonne sur la différence entre deux inerties on admettra que les causes externes d'erreurs sont éliminées : en particulier toute tendance liée au terrain survolé. Nous adjoindrons cependant les rafales de vents d'occurrence aléatoire. Il est alors naturel de supposer que  $\mathcal{E}(t)$  est de la forme :  $\mathcal{E}(t) = B(\langle \mathcal{E} \rangle_t)$ , où  $B(t)$  désigne un processus Brownien bidimensionnel et  $\langle \mathcal{E} \rangle_t$  la variation quadratique de  $\mathcal{E}(t)$ , l'analyse de la performance des centrales inertielles montrant que  $\langle \mathcal{E} \rangle_t \# 2q^2t$ .

La détermination de  $P(t)$  est menée dans l'annexe. En voici le résultat :

$$P(t) = \left[ \left( 2G\left(\frac{d}{q^2t}\right) - 1 \right) + \sqrt{\frac{2}{\pi}} \frac{q^2t}{d} e^{-\frac{d^2}{q^2t}} \right] \left[ G\left(\frac{l_1}{q^2t}\right) - G\left(\frac{l_2}{q^2t}\right) \right]$$

$$\text{avec } \bar{d} = d - E_n(0)$$

$$\bar{l}_1 = \frac{l}{2} - E_t(0)$$

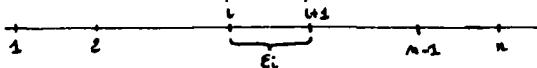
$$\bar{l}_2 = -\left(\frac{l}{2} + E_t(0)\right)$$

$$G(x) = \frac{1}{\sqrt{2\pi}} \int_{-\infty}^x e^{-\frac{u^2}{2}} du$$

$$\text{N.B. : si } q\sqrt{t}/d \gg 1, \text{ alors } P(t) \# \frac{1}{\pi} \frac{\bar{l} - 2E_t(0)}{\bar{d} - E_n(0)}$$

3 - PROBABILITES DE COLLISIONS DE  $n$  MISSILES SUR L'INTERVALLE  $(0, t)$ 

Dans le cadre de cette étude nous nous en tiendrons au vol groupé de  $n$  missiles disposés de front :



Nous simplifions l'analyse en ne considérant que des mouvements relatifs le long de la ligne de front. De cette manière la collision a nécessairement lieu entre deux missiles adjacents. Les missiles  $i$  et  $i+1$  entrent en collision si l'écart relatif  $\epsilon_i$  passe par le domaine  $\mathcal{E}_i$  entre deux recalages consécutifs. La probabilité de l'événement " $\epsilon_i$  passe par  $\mathcal{E}_i$ " a été établie dans le second chapitre.

Nous traitons d'abord le vol en formation entre deux recalages consécutifs, qui constituent autant de réinitialisation de la configuration. Nous admettons qu'à un instant précis une seule collision entre deux missiles peut survenir ; nous excluons les collisions triples ou d'un ordre supérieur, ces collisions étant beaucoup moins probables que des collisions simples.

Soit  $i(\omega)$  le nombre de missiles opérationnels à l'instant  $t$  ; il s'agit d'un processus stochastique à états discontinus. Il est clair que ce processus est markovien dans la mesure où le passé est résumé dans l'état courant. Considérons alors les probabilités de passage,  $P_{ij}(s, t)$ , probabilités que le système à l'état  $i$  à l'instant  $s$ , soit dans l'état  $j$  à l'instant  $t$ . Comme le processus est markovien, les diverses probabilités vérifient l'équation de Kolmogorov-Chapman :

$$P_{ij}(s, t) = \sum_{k \in \{0, \dots, n\}} P_{ik}(s, u) P_{kj}(u, t)$$

Il est usuel d'introduire les quantités :

$$a_{ij}(s) = \lim_{h \rightarrow 0} \frac{P_{ij}(s+h, s) - \delta_{ij}}{h}$$

L'analyse classique des processus markoviens conduit à l'équation différentielle :

$$\frac{\partial}{\partial t} P_{ij}(s, t) = \sum_{k \in \{0, \dots, n\}} P_{ik}(s, t) a_{kj}(t) \quad t > s$$

Partant de l'état  $i$  à l'instant  $t$  le seul changement possible est le passage de  $i$  à  $i-2$ . Par conséquent les deux seuls coefficients non nuls sont  $a_{ii}$  et  $a_{i-2}$ . Il est clair par ailleurs que

$$a_{ii} + a_{i-2} = 0$$

Posons

$$a_{i, i-2} = \epsilon_i(t)$$

L'équation différentielle se réduit à  $\frac{\partial}{\partial t} P_{ij}(s, t) = -\epsilon_i(t) P_{ij}(s, t) + \epsilon_{i-2}(t) P_{i-2, j}(s, t)$

$a_{i, i-2}(t)$  représente la densité de probabilité que le système passe de l'état  $i$  à l'état  $i-2$  à l'instant  $t$ . Puisque nous supposons que seule une collision simple éventuellement survient instantanément,  $a_{i, i-2}(t)$  est donc la densité de probabilité qu'il advienne à l'instant  $t$  une collision entre deux missiles adjacents.

A ce stade du raisonnement nous produisons un calcul qui n'est pas rigoureux, faute de pouvoir le traiter rigoureusement d'une manière simple. ( $i-1$ ) collisions sont possibles. Nous les considérons comme également probables et prenons pour probabilité d'occurrence de l'une d'entre elles la probabilité calculée au premier chapitre. En toute rigueur ce que nous venons de dire est faux. En effet la collision entre les missiles  $i$  et  $i+1$  provoque une scission de la famille des missiles en deux sous-familles : en particulier la distance entre les missiles  $i-1$  et  $i+2$ , à présent adjacents, sera plus élevée, en loi, que la distance entre missiles adjacents pris dans une même sous-famille.

La probabilité de collision  $P(t)$  entre deux missiles sur  $(0, t)$ , donc en fait à l'instant  $t$ , a été calculée au premier chapitre.

$$\text{Soit } a(t) = \frac{d}{dt} P(t)$$

D'après ce que nous venons de dire :

$$a_{i, i-2}(t) = (i-1) a(t)$$

Il est utile de distinguer les deux cas,  $n$  pair et  $n$  impair. Par ailleurs nous ferons systématiquement  $s = 0$  et poserons :  $P_{ik}(t) = P_{ik}(0, t)$

1er cas :  $n$  pair -  $n = 2m$  ( $m > 1$ )

L'équation différentielle s'écrit encore :

$$\frac{d}{dt} \begin{vmatrix} P_{2m, 0} \\ P_{2m, 2} \\ \vdots \\ P_{2m, 2m-2} \\ P_{2m, 2m} \end{vmatrix} = \begin{vmatrix} 0 & \epsilon_2 & & & P_{2m, 0} \\ 0 & -\epsilon_2 & \epsilon_4 & & P_{2m, 2} \\ & & (0) & -P_{2m-2} & P_{2m, 2m-2} \\ & & & -P_{2m} & P_{2m, 2m} \end{vmatrix}$$

$$= a(t) \begin{vmatrix} 0 & 1 & & & P_{2m, 0} \\ 0 & -1 & 3 & & P_{2m, 2} \\ & & (0) & -2m-3 & P_{2m, 2m-2} \\ & & & -2m-1 & P_{2m, 2m} \end{vmatrix}$$

2<sup>ème</sup> cas : n impair n = 2m+1 (m > 1)

$$\frac{d}{dt} \begin{vmatrix} p_{m+1,1} \\ p_{m+1,3} \\ \vdots \\ p_{m+1,2m-1} \\ p_{m+1,2m+1} \end{vmatrix} = \begin{vmatrix} 0 & e_3 & & & p_{m+1,1} \\ 0 & -e_3 & e_3 & \dots & p_{m+1,3} \\ & & (0) & & \\ & & -e_{2m-2} & e_{2m+1} & p_{m+1,2m-1} \\ & & -e_{2m+2} & & p_{m+1,2m+1} \end{vmatrix}$$

$$= a(t) \begin{vmatrix} 0 & 2 & & & p_{m+1,1} \\ 0 & -2 & 4 & \dots & p_{m+1,3} \\ & (0) & & & \\ & -2(m-1) & 2m & & p_{m+1,2m-1} \\ & & 2m & & p_{m+1,2m+1} \end{vmatrix}$$

Les conditions initiales sont respectivement :

$$p_{m+1,2m}(0) = 1 \quad p_{m+1,j}(0) = 0 \quad j \neq 2m$$

$$p_{m+1,2m+1}(0) = 1 \quad p_{m+1,j}(0) = 0 \quad j \neq 2m+1$$

La structure très particulière de la matrice dynamique rend la résolution immédiate. Sa base de mise sous forme d'une matrice de Jordan reste fixe au cours du temps. Ainsi la matrice commute avec sa dérivée. Si on l'écrit :  $a(t) A_0$  alors, en notant :

$$P(t) = (p_1, p_{2m}, \dots)^*(t)$$

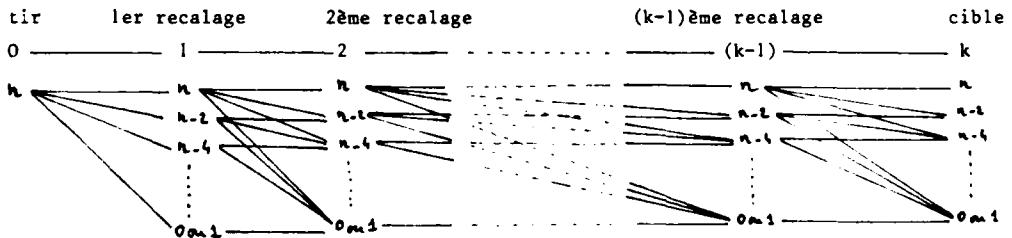
$$P(t) = \exp \left\{ \left( \int_0^t a(s) ds \right) A_0 \right\} p_0 \quad \text{or} \quad \int_0^t a(s) ds = P(t)$$

il vient :

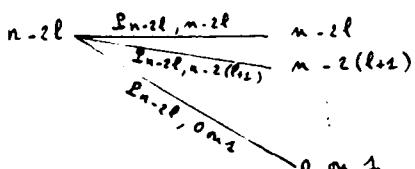
$$P(t) = \exp \left\{ P(t) A_0 \right\} \begin{vmatrix} 0 \\ \vdots \\ 1 \end{vmatrix}$$

#### 4 - CAS GENERAL : SALVE DE n MISSILES, TRAJECTOIRE EN K TRONÇONS

Interessons-nous à présent à une mission de n missiles volant de front dans un même plan horizontal. La trajectoire est découpée en k intervalles de même durée auxquels sont associés (k-1) recalages. Les états successifs possibles sont disposés selon le graphe planaire ci-dessous. Puisque nous avons fait l'hypothèse que les collisions se succèdent sans coïncidence entre deux missiles adjacents, les états probables se déduisent de n selon la progression arithmétique n, n-2, n-4, ..., 0



Etats accessibles à partir de l'état n - 2l



Les évolutions chronologiques (description horizontale) sont indépendantes entre elles parce que chaque recalage altimétrique constitue une nouvelle initialisation de la configuration spatiale, tandis que les évolutions synchrones (description verticale) sont exclusives les unes des autres. Les états finaux accessibles sont :

$$\mathcal{V} \in \{n, n-2, n-4, \dots, 0m1\}$$

et les probabilités qui y sont attachées sont notées  $\pi_{\mathcal{V}}$ . Elles s'obtiennent par calcul combinatoire. Ainsi  $\pi_n$  est la probabilité qu'il n'y ait eu aucune collision sur les k tronçons :

$$\pi_n = (\pi_{m, m})^k = \exp(-k(m-1) P(t))$$

La probabilité  $\pi_{n-2}$  d'une seule collision s'exprime par :

$$\pi_{n-2} = \sum_{j=1}^k (P_{n,n})^{j-1} \times P_{n,n-2} \times (P_{n-2,n-2})^{k-j}$$

(j-1) tronçons  
sans collision      collision au  
jème tronçon      (k-j) tronçons  
sans collision

comme :

$$P_{n,n} = \exp(-(\alpha-1) \bar{P}(t))$$

$$P_{n-2,n-2} = \exp(-(\alpha-3) \bar{P}(t))$$

Il vient :

$$\pi_{n-2} = P_{n,n-2} \exp(-(\alpha-1)(\alpha-3) \bar{P}(t)) \frac{1 - \exp(-2k \bar{P}(t))}{1 - \exp(-2 \bar{P}(t))}$$

Explicitons encore la probabilité  $\pi_{n-4}$ . Deux voies peuvent être empruntées pour aboutir à l'état final ( $n-4$ ). Il surviendra deux collisions simples sur le même intervalle ou sur deux intervalles distincts. Par conséquent :

$$\pi_{n-4} = \sum_{j=1}^k (P_{n,n})^{j-1} P_{n,n-4} (P_{n-4,n-4})^{k-j} +$$

$$\sum_{j=1}^k (P_{n,n})^{j-1} P_{n,n-2} \left( \sum_{l=1}^{k-j} (P_{n-2,n-2})^{l-1} P_{n-2,n-4} (P_{n-4,n-4})^{k-j-l} \right)$$

Tous calculs faits il reste :

$$\pi_{n-4} = P_{n,n-4} e^{-(\alpha-1)(\alpha-5) \bar{P}(t)} \frac{1 - e^{-3k \bar{P}(t)}}{1 - e^{-3 \bar{P}(t)}}$$

$$+ P_{n,n-2} P_{n-2,n-4} \frac{e^{-(\alpha-2)(\alpha-5) \bar{P}(t)}}{1 - e^{-2 \bar{P}(t)}} \left[ \frac{1}{1 - e^{-4 \bar{P}(t)}} - \frac{e^{-2(\alpha-2) \bar{P}(t)}}{1 - e^{-2 \bar{P}(t)}} \right]$$

Les calculs deviennent ensuite de plus en plus fastidieux à développer aussi arrêterons-nous ici d'expliquer les valeurs.

Le nombre moyen de missiles à l'issue du parcours total, à l'instant d'atteindre la cible, vaut :

$$N(n,k) = \sum \nu \pi_\nu$$

et le taux moyen de perte est défini par :

$$\rho(n,k) = \frac{n - N(n,k)}{n} = 1 - \pi_n - (1 - \frac{2}{n}) \pi_{n-2} - \dots$$

### 5 - QUELQUES CAS NUMERIQUES

A titre d'exemple nous traiterons les trois cas  $n = 2, 5$  et  $6$  avec par ailleurs  $k = 15$ . Le coefficient réduit  $d/\sqrt{2}q^2t$  sera supposé compris entre  $0.250$  et  $0.500$ .

La longueur  $\frac{1}{2}t$  est ici fixée à  $25 \cdot 10^{-4}$ . En négligeant les conditions initiales après recalage :

$d/\sqrt{2}q^2t$	$d/2q^2t$	$p(t)$
0.250	3,3 $10^{-2}$	
0.375	2,2 $10^{-2}$	
0.500	1,6 $10^{-2}$	

Cas  $n = 2$

$d/\sqrt{2}q^2t$	$P_{2,2}$	$P_{2,0}$	$\pi_2$	$\pi_0$	$\rho = \pi_0$
0,250	0,967	0,033	0,6045	0,3955	40 %
0,375	0,978	0,022	0,7163	0,2837	28 %
0,500	0,984	0,016	0,7851	0,2149	21 %

Le taux moyen de perte est égal dans ce cas particulier à  $\pi_0$ .

Cas  $n = 5$

Il conduit à la matrice :

$$A_0 = \begin{vmatrix} 0 & 2 & 0 \\ 0 & -2 & 4 \\ 0 & 0 & -4 \end{vmatrix}$$

La diagonalisation permet d'écrire :

$$A_0 = \begin{vmatrix} 1 & -1 & -1 \\ 0 & 1 & 2 \\ 0 & 0 & -1 \end{vmatrix} \begin{vmatrix} 0 & 0 & 0 \\ 0 & -2 & 0 \\ 0 & 0 & -4 \end{vmatrix} \begin{vmatrix} 1 & 1 & 1 \\ 0 & 1 & 2 \\ 0 & 0 & -1 \end{vmatrix}$$

Nous en déduisons :

$$P_{5,1} = 1 - 2e^{-2p} + e^{-4p}$$

$$P_{5,3} = 2(e^{-2p} - e^{-4p})$$

$$P_{5,5} = e^{-4p}$$

D'où le tableau :

$d/\sqrt{2q^2t}$	$P_{5,1}$	$P_{5,3}$	$P_{5,5}$	$\pi_5$	$\pi_3$	$\pi_1$	$\ell$
0,250	$4,08 \cdot 10^{-3}$	0,120	0,876	0,138	0,467	0,401	50 %
0,375	$1,85 \cdot 10^{-3}$	0,0824	0,916	0,267	0,499	0,233	39 %
0,500	$1,0 \cdot 10^{-3}$	0,0610	0,938	0,383	0,472	0,145	30 %

N.B. : Le taux moyen de perte vaut :  $\ell = 1 - \pi_5 - (1-\pi_5)\pi_3 - (1-\pi_5-\pi_3)\pi_1$

Cas n = 6

Il conduit à la matrice :

$$A_0 = \begin{vmatrix} 0 & 1 & 0 & 0 \\ 0 & -1 & 3 & 0 \\ 0 & 0 & -3 & 5 \\ 0 & 0 & 0 & -5 \end{vmatrix}$$

La diagonalisation fournit :

$$A_0 = \begin{vmatrix} 1 & -1 & -1 & 3 \\ 0 & 1 & 3 & -15 \\ 0 & 0 & -2 & 20 \\ 0 & 0 & 0 & -8 \end{vmatrix} \begin{vmatrix} 0 & 0 & 0 & 0 \\ 0 & -1 & 0 & 0 \\ 0 & 0 & -3 & 0 \\ 0 & 0 & 0 & -5 \end{vmatrix} \begin{vmatrix} 1 & 1 & 1 & 1 \\ 0 & 1 & 3/2 & 15/8 \\ 0 & 0 & -1/2 & -10/8 \\ 0 & 0 & 0 & -1/8 \end{vmatrix}$$

Il s'en suit que :

$$\varrho_{6,0} = 1 - \frac{15}{8} e^{-\ell(t)} + \frac{19}{8} e^{-3\ell(t)} - \frac{3}{8} e^{-5\ell(t)}$$

$$\varrho_{6,2} = \frac{15}{8} e^{-\ell(t)} - \frac{30}{8} e^{-3\ell(t)} + \frac{15}{8} e^{-5\ell(t)}$$

$$\varrho_{6,4} = \frac{20}{8} (e^{-3\ell(t)} - e^{-5\ell(t)})$$

$$\varrho_{6,6} = e^{-5\ell(t)}$$

D'où le tableau :

$d/\sqrt{2q^2t}$	$P_{6,0}$	$P_{6,2}$	$P_{6,4}$	$P_{6,6}$	6	4	2	0	$\ell$
0,250	$8,34 \cdot 10^{-5}$	$7,40 \cdot 10^{-3}$	0,145	0,848	0,0842	0,356	0,0937	0,466	65 %
0,375	$2,53 \cdot 10^{-5}$	$3,40 \cdot 10^{-3}$	0,101	0,896	0,192	0,449	0,0055	0,354	51 %
0,500	$9,88 \cdot 10^{-6}$	$1,83 \cdot 10^{-3}$	0,075	0,923	0,301	0,464	# 0	0,235	39 %

N.B. : le taux moyen de perte est  $\ell = 1 - \pi_6 - \frac{2}{3}\pi_4 - \frac{1}{3}\pi_2$

## 6 - CONCLUSION

1 - La détermination du taux moyen de perte s'effectue en plusieurs étapes. La première est relative à la collision de deux missiles qui volent de conserve et de front dans un même plan horizontal. La probabilité est égale au nombre moyen de passages par un certain domaine de dimension 1 d'un processus bidimensionnel qui représente essentiellement la dérive relative des inerties propres à chaque missile. Elle dépend de  $d$ , écart transverse de consigne entre deux missiles, de  $l$  plus petit espace entre deux missiles, de la durée  $t$  du trajet entre deux recalages consécutifs et des erreurs relatives de recalage.

Lors du vol de plusieurs missiles sur une même ligne de front nous admettons qu'à un instant précis seule une collision simple entre deux missiles adjacents peut survenir. La loi qui donne le nombre de missiles subsistant au  $(n+1)$ ème recalage en fonction du nombre de missiles au  $n$ ème recalage est fournie par une équation différentielle de Kolmogoroff-Chapman. Celle-ci se résoud explicitement

Une mission de  $n$  missiles, tirés pour voler de front, comprend  $(k-1)$  recalages auxquels correspondent  $k$  tronçons de trajectoire que l'on suppose survolés dans le même temps  $t$ . La probabilité qu'il subsiste  $i$  missiles à l'arrivée sur la cible se déduit des résultats précédents par une analyse combinatoire menée au chapitre n° 4).

2 - Rappelons à présent le taux moyen de perte calculé dans les trois cas  $n = 2, 5$  et  $6$  et avec les hypothèses :

$$k = 15$$

$$\frac{1}{\sqrt{2q^2t}} = 25 \cdot 10^{-3}$$

$$0,250 \leq d/\sqrt{2q^2t} \leq 0,500$$

$n$	$d/\sqrt{2q^2t}$	0,250	0,375	0,500
2		40 %	28 %	21 %
5		50 %	39 %	30 %
6		65 %	51 %	39 %

## ANNEXE : Loi de récurrence du mouvement Brownien dans le plan

Considérons un processus de Wiener (ou mouvement Brownien) normalisé dans le plan  $\mathbb{R}^2$  :  
 $B(s) = (B_1(s), B_2(s))$

Pour déterminer le nombre de fois que  $B(s)$  passe par  $(\mathcal{E} : x = d, l_2 \leq y \leq l_1)$  dans l'intervalle de temps  $[0, t]$  nous considérons la fonction :

$$n(t, d) = \int_0^t \mathbb{I}(d \leq B_1(s) \leq d + \Delta x; l_2 \leq B_2(s) \leq l_1) ds \Delta x^2 = \frac{d}{\Delta x} \int_0^t \mathbb{I}(B_1(s) \leq d + \Delta x; l_2 \leq B_2(s) \leq l_1) ds$$

Afin de résoudre de manière simple et élégante le problème qui nous occupe nous faisons appel à des éléments d'Analyse Non Standard.

Soit  $X$  le relèvement de  $B_1$  dans  $\mathbb{R}$ , version non standard de l'espace réel ; à l'élément différentiel  $\Delta x^{-1}$  est associé l'infiniment petit  $\Delta t^{1/2}$ . Alors le relèvement de  $n$  s'écrit :

$$N(t, d) = \sum_{0 \leq s \leq t} \mathbb{I}(B_1(s) = d; l_2 \leq B_2(s) \leq l_1) \Delta t^{1/2}$$

Introduisons :

$$\text{Montrons que : } |Y(t) - d| = \int_0^t \text{signe}(B_1(s) - d) \mathbb{I}(l_2 \leq B_2(s) \leq l_1) dB_2(s) + n(t, d)$$

Il suffit de le vérifier pour  $d = 0$  et sur les relèvements non standard.

$$|Y(t + \Delta t) - Y(t)| = \begin{cases} \Delta x(t) & \text{si } x(t) > 0 \text{ et } l_2 \leq B_2(t) \leq l_1 \\ \Delta t^{1/2} & \text{si } x(t) = 0 \text{ et } l_2 \leq B_2(t) \leq l_1 \\ -\Delta x(t) & \text{si } x(t) < 0 \text{ et } l_2 \leq B_2(t) \leq l_1 \\ 0 & \text{si } B_2(t) \leq l_2 \text{ ou } B_2(t) \geq l_1 \end{cases}$$

D'où

$$|Y(t)| = \sum_{0 \leq s \leq t} \text{signe}(x(s)) \mathbb{I}(l_2 \leq B_2(s) \leq l_1) \Delta x(s) + \sum_{0 \leq s \leq t} \mathbb{I}(x(s) = 0 \text{ et } l_2 \leq B_2(s) \leq l_1) \Delta t^{1/2}$$

$$\text{Il vient } |Y(t)| = \int_0^t \text{signe}(x(s)) \mathbb{I}(l_2 \leq B_2(s) \leq l_1) dx(s) - N(t, 0)$$

L'ombre sur l'espace réel usuel fournit :

$$|Y(t)| = \int_0^t \text{signe}(B_1(s)) \mathbb{I}(l_2 \leq B_2(s) \leq l_1) dB_2(s) + n(t, 0)$$

et plus généralement :

$$|Y(t) - d| = \int_0^t \text{signe}(B_1(s) - d) \mathbb{I}(l_2 \leq B_2(s) \leq l_1) dB_2(s) + n(t, d)$$

Nous déduisons de cette remarquable identité que  $n(t, d)$  est lui-même un processus stochastique qui admet pour espérance mathématique :  $E(n(t, d)) = E(|Y(t) - d|)$

soit encore d'après la définition de  $Y(t)$  :

$$E(n(t, d)) = E(|B_1(t) - d|) \mathbb{P}(l_2 \leq B_2(t) \leq l_1)$$

$$E(|B_1(t) - d|) = \frac{1}{\sqrt{2\pi}} \int_{-\infty}^{\infty} |u - d| e^{-\frac{u^2}{2}} du = \frac{1}{\sqrt{2\pi}} \left( \int_{-\infty}^d (d-u) e^{-\frac{u^2}{2}} du + \int_d^{\infty} (u-d) e^{-\frac{u^2}{2}} du \right) \\ = d \left( 2G\left(\frac{d}{\sqrt{2\pi}}\right) - 1 \right) + \sqrt{\frac{2}{\pi}} e^{-\frac{d^2}{2}}$$

$$\text{avec } G(u) = \frac{1}{\sqrt{2\pi}} \int_{-\infty}^u e^{-\frac{v^2}{2}} dv$$

$$\text{En définitive } E(n(t, d)) = \left[ 2d \left( G\left(\frac{d}{\sqrt{2\pi}}\right) - 1 \right) + \sqrt{\frac{2}{\pi}} e^{-\frac{d^2}{2}} \right] \left[ G\left(\frac{l_1}{\sqrt{2\pi}}\right) - G\left(\frac{l_2}{\sqrt{2\pi}}\right) \right]$$

Si à présent le processus est homogène à une longueur il existe une unique manière de définir un processus homogène pour lequel  $n(t, d)$  représente le nombre de passage par  $\mathcal{E}$ .

Soit le temps  $\tau$  tel que :

$$d^2 = q^2 \tau$$

le passage aux variables réduites s'effectue comme suit :

$$x(\text{longueur}) \rightarrow \frac{x}{d}$$

$$s(\text{temps}) \rightarrow \frac{s}{\tau}$$

$$\text{alors } n(t, x) \text{ est défini par } n(t, x) = \int_0^t \mathbb{I}(x \leq B_1(s) \leq x + \Delta x) \frac{ds}{\tau} d(\Delta x)^2$$

Exprimé en termes d'Analyse non Standard

$$\Delta x \rightarrow q \Delta t^{1/2}$$

$$ds \rightarrow \Delta t$$

$$\text{L'élément différentiel devient : } \frac{\Delta t}{\tau} d \frac{q}{d} q^{-1} (\Delta t)^{-1/2} = \frac{d}{q\tau} \Delta t^{1/2} = \frac{\Delta t}{\tau^{1/2}}$$

Tous calculs effectués, il vient :

$$E(n(t, x)) = \left[ 2 \left( G\left(\frac{d}{q\tau^{1/2}}\right) - 1 \right) + \sqrt{\frac{2}{\pi}} \frac{q}{d} e^{-\frac{d^2}{2q^2\tau}} \right] \left[ G\left(\frac{l_1}{q\tau^{1/2}}\right) - G\left(\frac{l_2}{q\tau^{1/2}}\right) \right]$$

Si le domaine  $\mathcal{E}$  est défini par  $x = d$ ,  $-\frac{\ell}{2} \leq y \leq \frac{\ell}{2}$  et si les conditions initiales de  $(B(t))$  sont

$$x_0 = B_1(0)$$

$$y_0 = B_2(0)$$

On remplace dans la précédente formule  $d$  par  $d - B_1(0)$ ,  $\ell_1$  par  $\frac{\ell}{2} - B_2(0)$  et  $\ell_2$  par  $-(\frac{\ell}{2} + B_2(0))$

**APPLICATIONS DES CENTRALES A COMPOSANTS LIÉS  
AUX MISSILES TACTIQUES :  
CAS DES MISSILES STAND-OFF**



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Les Centrales à Composants Liés sont particulièrement bien adaptées aux missiles tactiques stand-off. La certitude en a été acquise après plusieurs années d'études théoriques et d'essais. Les études théoriques préliminaires ont utilisé largement les paramètres définissant en détail les trajectoires des missiles et leurs sollicitations déduits des mesures faites sur les systèmes d'armes alors en développement ou en production. Les senseurs et principalement les gyroscopes ont été testés en laboratoire. Ces études préliminaires ont abouti aux choix de différentes technologies (ex : gyrolaser et gyromécanique) adaptées chacune au système d'arme concerné. La répartition des fonctions entre les différents équipements a été définie. La comparaison des avantages et inconvénients de chacune des technologies montre qu'aujourd'hui le gyrolaser est plutôt réservé aux grandes et moyennes précisions, les gyromécaniques (à toupie) aux moyennes et faibles précisions. Les exigences concernant les performances dynamiques et la robustesse sont de plus en plus contraignantes. C'est dans ces domaines que les progrès sont les plus attendus.

## INTRODUCTION

Jusqu'au début des années 70, les Systèmes de Navigation Inertielle, le plus souvent constitués par des plateformes stabilisées, étaient réservés aux missiles tactiques complexes de grande portée et grande précision. L'arrivée sur le marché des Centrales à Composants Liés (CCL) a constitué une véritable révolution technique sur les prix, volumes et fiabilité, ouvrant le champ d'application des Systèmes Inertiels à une très large gamme de missiles tactiques, les missiles stand-off en étant les premiers bénéficiaires pour nos propres développements.

Après plusieurs années d'études et d'essai, les résultats acquis sont tout-à-fait positifs : la technologie CCL a bien confirmé les espérances sur les points déjà cités, mais encore propose une si grande diversité de matériels qu'elle permet mieux que les solutions classiques d'approcher la solution optimale. Il est vrai que si la solution obtenue est bien appropriée, plus simple et moins chère, c'est au prix d'un accroissement des travaux théoriques et d'une complication des essais qui grèvent les coûts de la phase de développement.

Cet article présente dans une première partie le domaine d'applications traité, dans une deuxième partie les travaux préliminaires permettant d'établir la faisabilité de cette technique pour ce domaine d'applications, et en troisième partie les critères et méthodes à respecter pour faire un choix parmi les technologies disponibles. La quatrième partie décrit les procédures d'évaluation retenues. Enfin, après avoir présenté quelques éléments de comparaison entre les systèmes existants, l'article envisage les améliorations prévisibles grâce aux technologies nouvelles qui sont en cours de développement.

## DOMAINE D'APPLICATION

Les missiles stand-off sont d'une grande diversité sur le plan de leur mission, de leurs dimensions de leurs performances. Les paramètres qui dimensionnent les Centrales à Composants Liés (CCL) peuvent varier d'une manière plus ou moins corrélée les unes aux autres dans les rapports suivants :

Portée	1 à 100
Temps de vol	1 à 100
Vitesse	1 à 5
Temps de réaction	1 à 100
Etendues de mesure	1 à 5
Perturbations	1 à 30 (vibrations linéaires, angulaires)
Précisions	1 à 300 (rapportées à la distance parcourue)
Classe de précision des senseurs	1 à 1000
Puissance de calcul	1 à 50

Ces différences considérables entraînent des différences (moins considérables) des techniques employées. Cependant, les méthodes d'étude et essais ont un caractère assez général comme la suite tente de le montrer.

## TRAVAUX PRÉLIMINAIRES

Une révolution technique est toujours accueillie par les techniciens avec grand intérêt (possibilités nouvelles) mais avec prudence (risques). Chaque domaine présente des difficultés particulières. Les Systèmes Inertiels en présentent un grand nombre. L'une d'entre elles est l'impossibilité de valider le système en dehors de son emploi réel. En effet, les entrées sont des accélérations linéaires et des vitesses angulaires qu'il faut connaître pour dimensionner le système et qu'il faut réaliser pour le valider. Or au stade du projet, les caractéristiques des petits mouvements et vibrations ne peuvent pas être connues avec le détail nécessaire (corrélations d'amplitude et de phase en particulier) et la réalisation du profil des accélérations ne peut être faite que sur une trajectoire réelle. Il y a donc un risque que le vol du missile modifie sensiblement les résultats acquis par ailleurs. Les CCL sont d'autant plus sensibles à ces phénomènes qu'elles ne sont pas découpées des mouvements du missile.

Les travaux d'étude pour établir la faisabilité de la technologie CCL ont concerné les senseurs (essentiellement les gyroscopes), les algorithmes de traitement et les calculateurs.

### ETUDES THÉORIQUES

Elles avaient deux objectifs : définir approximativement les performances de la CCL pour les applications envisagées et définir les logiciels de navigation.

Il a fallu tout d'abord constituer un outil de travail qui nous permette d'évaluer les solutions imaginées et d'en dégager les avantages et les inconvénients. Il s'agit de connaître les informations qui constituent les entrées du système inertiel, c'est-à-dire les caractéristiques des mouvements du missile en vol, que ce soient les paramètres définissant sa trajectoire ou les petits mouvements autour de celle-ci.

Ces travaux ont utilisé largement les résultats des mesures faites sur les missiles qui étaient alors en développement et en production. Les radars et moyens optiques des champs de tir caractérisent la trajectoire, les attitudes, accélérations et vibrations sont déduites des mesures internes du missile transmises par la télémesure. Il en a été déduit :

- a) Plusieurs trajectoires caractéristiques définies par les accélérations et vitesses angulaires échantillonées à cadence très rapide
- b) Les répartitions spectrales des accélérations linéaires et des vitesses angulaires suivant les différentes phases du vol.

Cet outil a participé à trois études : l'évaluation des erreurs de navigation déduites des modèles théoriques du comportement des senseurs, l'évaluation de l'influence des erreurs d'initialisation et la définition des logiciels de navigation. La démarche est représentée sur la fig. 1.

La première tâche est de définir une trajectoire de référence déduite des fichiers par un calcul très précis : intégration par la méthode de Runge-Kutta d'ordre 4, en double précision, à cadence rapide.

Les erreurs des senseurs ont été définies par des études bibliographiques et par des contacts avec le LRBA (Laboratoire de Recherche Balistique et Aérodynamique) et les industriels fabriquant les senseurs. Ces erreurs sont introduites au niveau des fichiers définissant les trajectoires. Les erreurs de navigation sont obtenues par comparaison avec la trajectoire de référence. Cette étude nous a permis de rédiger la version initiale des spécifications techniques des senseurs.

Les études d'erreurs d'initialisation ne nécessitent pas de développement.

L'étude des logiciels a porté sur les points suivants : le choix de l'opérateur de rotation, l'ordre d'intégration des vitesses angulaires, les cadences de calcul, la valeur du LSB et la précision des opérations mathématiques.

L'opérateur de rotation a été choisi indépendamment des trajectoires à réaliser. La méthode des quaternions a été préférée à celles des cosinus directeurs ou des angles d'Euler. C'est d'ailleurs la solution retenue par de nombreux utilisateurs. Le quaternion est en effet bien adapté au calcul arithmétique en temps réel car il n'utilise pas de lignes trigonométriques et conserve l'orthogonalité des trièdres. Il n'a pas de position interdite.

Les autres éléments ont été obtenus à partir des trajectoires simulées. Les résultats globaux sont les suivants :

- L'ordre d'intégration est en général égal à 3. Il peut être supérieur pour certaines applications où les variations des grandeurs d'entrée sont très rapides. On choisit en effet à performances égales, la solution qui conduit à un algorithme plus complexe associé à une cadence plus lente.
- Les calculs sont faits à deux cadences dans un rapport 10 (environ) l'une de l'autre. Le calcul des attitudes et des vitesses est fait à la cadence la plus élevée, les calculs de position, correction de navigation terrestre et normalisation du quaternion à la cadence la plus lente.
- La quantification des informations nécessaires au pilotage est plus contraignante que celle nécessaires à la navigation. Elles sont donc définies par ailleurs.

Les algorithmes ont été d'autre part vérifiés en présence de vibrations et rotations à valeur moyenne nulle (mouvements coniques en particulier) pour en évaluer la robustesse vis-à-vis de ce type de perturbations.

#### ETUDE DES SENSEURS

Une veille technologique permanente permet de déceler les technologies prometteuses quelques années avant leur arrivée sur le marché. Les accéléromètres étant déjà disponibles, l'effort de veille a porté principalement sur les gyroscopes. Les performances exigées sont inhabituelles pour un équipement de missile tactique ; en effet la dynamique du senseur qui est l'étendue de mesure divisée par l'erreur de zéro atteint  $10^5$  pour les performances moyennes et dépasse  $10^8$  pour les performances élevées.

Les fabricants de gyroscopes ont mis à notre disposition des maquettes réputées tenir cette performance. Il a fallu tout d'abord mettre au point dans nos laboratoires les matériels, procédures et traitements capables de les vérifier. Les travaux ont porté à la fois sur des gyroscopes mécaniques et des gyrolasers. Les modèles d'erreur adaptés à l'utilisation missile tactique ont été affinés et introduits sur les trajectoires obtenues au cours de l'étude théorique. Les spécifications techniques détaillées des senseurs en ont été déduites pour chaque application envisagée.

Le tableau 1 présente les principaux paramètres faisant l'objet d'une spécification et le type de traitement retenu : le paramètre peut être non traité ou compensé. La compensation consiste à déduire des équations du modèle l'influence des paramètres pris en compte et dont les valeurs ont été préalablement mémorisées. Certaines de ces valeurs varient avec le temps et doivent être remises à jour périodiquement : c'est la calibration. Le tableau est établi pour les trois types de senseurs les plus utilisés : le gyrolaser, le gyroscope à suspension élastique accordée, l'accéléromètre pendulaire asservi et pour deux classes d'application : grande précision et moyenne précision.

## LES CALCULATEURS

Ils ont constitué longtemps l'obstacle principal à la technologie CCL. Les progrès considérables faits dans les années 70 ont levé cette barrière. Les études ont porté sur les structures des machines, les composants de base, les langages.

## RESULTATS

Les travaux préliminaires ont établi la faisabilité de la technologie CCL pour la plupart des missiles tactiques et parmi ceux-ci les missiles stand-off, excepté provisoirement les missiles les plus précis. Ils ont aussi mis en évidence les caractéristiques les plus contraignantes. Elles doivent être impérativement prises en compte au cours des études de guidage et pilotage ultérieures.

## **CRITÈRES ET MÉTHODES DE CHOIX**

La faisabilité des CCL étant acquise, cette technologie a trouvé des applications immédiates. Il fallait faire des choix parmi les solutions envisagées. Ce chapitre présente en première partie les critères retenus, en deuxième partie les caractéristiques générales des solutions choisies et traite enfin deux points qui font l'objet de compromis difficiles.

### CRITERES

Les critères de choix ne sont pas essentiellement différents de ceux de tout équipement électronique. Le poids relatif attribué à chacun d'entre eux dépend de l'utilisation. Le classement proposé ici a un caractère indicatif étant donnée la diversité des missiles stand-off.

Le premier critère concerne le volume : malgré les progrès considérables faits dans ce domaine, il reste une contrainte de première importance. La comparaison des caractéristiques des missiles anti-navire de première et deuxième génération est significative (Tab. N° 2). Il faut noter que les portées des plus récents sont 3 à 4 fois supérieures.

Bien que la part occupée par le système de guidage pilotage (SGP) soit minime par rapport à l'ensemble du missile, il est avantageux de la réduire encore quand on sait que par exemple 1 cm de longueur économisé représente un gain de 1 km en portée.

Un deuxième critère d'importance capitale concerne les contraintes apportées par le système de guidage et de pilotage du missile à son lanceur. Que ce soit un véhicule terrestre, un hélicoptère, un navire ou un avion, il doit conserver le maximum de ses capacités opérationnelles dans la phase préparatoire au tir pendant laquelle le Système Inertiel est initialisé. Les éléments à prendre en compte sont entre autres : la durée des phases préalables à la mise en route (préchauffage), la durée de l'initialisation, les limitations des évolutions du porteur ou au contraire, les trajectoires imposées.

Le troisième critère concerne la capacité du matériel à supporter les conditions de stockage et d'emploi opérationnel du Système d'Armes. Malgré des durées de stockage de plusieurs années, le système doit être opérationnel sans autre opération de déstockage que son utilisation réelle. Les opérations de maintenance préventive doivent être rares, aisées, rapides. La situation idéale serait de pouvoir utiliser les missiles tactiques comme des munitions.

Les coûts sont aussi un élément fondamental des choix. Comme les volumes, mais en moindre proportion, les coûts des Systèmes inertIELS ont diminué sensiblement ces dernières années, la technologie CCL étant elle-même un élément déterminant de cette évolution. En outre, la complication croissante des Systèmes d'Armes diminue encore la part des Systèmes InertIELS. Il demeure que les CCL restent un élément significatif des prix, particulièrement pendant le développement.

Les performances peuvent sembler en bien mauvaise position dans cette énumération. C'est que les technologies parmi lesquelles le choix doit être fait sont toutes supposées remplir le cahier des charges techniques minimales. Le critère performances joue en effet en tout ou rien. Si elles ne sont pas tenues, le système est rejeté. Dans le cas où elles sont meilleures que les spécifications minimales, elles restent un avantage sérieux mais non primordial en permettant d'élargir les contraintes imposées à d'autres fonctions.

Le dernier critère cité ici concerne les capacités d'évolution des technologies retenues. Nous avons souligné plus haut la valeur élevée des coûts de développement. Ils sont plus rapidement amortis si une technologie déjà étudiée et validée par ailleurs peut être utilisée dans une application différente.

### CARACTERISTIQUES GENERALES DES SOLUTIONS RETENUES

Malgré la grande diversité des applications missiles tactiques stand-off et malgré les différences des poids attribués aux principaux critères dans chaque cas particulier, les solutions que nous avons retenues respectent quelques principes généraux. Ils sont rapidement présentés dans ce chapitre.

#### Structure générale :

L'élément de choix le plus caractéristique concerne la plus ou moins grande centralisation des fonctions de calcul du missile; celles-ci sont présentées dans le Tableau n° 3 qui est limité aux fonctions utilisant les mesures inertielles.

Dans la mesure du possible, c'est la structure décentralisée qui est retenue. Toutefois, les calculs peuvent être plus ou moins regroupés dans certaines applications où les contraintes de volume et de coût sont plus difficiles à respecter que les contraintes de performances. La discussion détaillée est traitée ultérieurement.

#### Echange des informations

Il est rarement possible d'utiliser un mode de transmission unique. Les informations les plus rapides destinées essentiellement au pilotage sont fournies sous forme analogique ou par l'intermédiaire d'un bus parallèle rapide, dont le nombre d'abonnés est limité. Les échanges plus lents utilisent un bus série standard au sens où il correspond si possible à ceux utilisés par les lanceurs. Ceci est représenté en figure 2.

#### Les senseurs :

Il n'est pas question de faire le choix d'une technologie unique. En effet chacune d'entre elles correspond à un certain domaine de performances avec des zones de recouvrement où par exemple des gyroscopes aussi différents que le gyrolaser ou le gyroscope à suspension élastique accordée se trouvent en concurrence. Il est souvent possible, dans la mesure des capacités financières des programmes, de faire vivre simultanément les technologies compatibles avec les exigences du système d'arme en situation de quasi transparence. Il faut noter cependant que certaines caractéristiques telles que les bruits ou le comportement dynamique sont spécifiques du type de senseurs ; il faut alors que le calculateur reconnaisse le senseur utilisé pour orienter le logiciel vers les modules adaptés à celui-ci.

#### Calculateurs :

Comme pour les senseurs, il n'est pas question de définir une structure et une technologie uniques. On peut cependant souligner l'intérêt que présentent les machines microprogrammables pour les calculs en temps réel à caractère répétitif : échanges, opérations mathématiques particulières...

### COMPROMIS A REALISER DANS LES CHOIX

Les critères de choix sont trop nombreux pour être tous également satisfaits. Ce paragraphe traite de deux compromis particuliers, rencontrés systématiquement pour les applications des CCL aux missiles tactiques stand-off. Il s'agit des contradictions entre les exigences performances-volume et entre les caractéristiques des informations nécessaires à la navigation et au pilotage.

C'est une évidence d'observer que les précisions d'un système inertiel croissent avec son volume. Il est plus discutable mais généralement vrai que les performances dynamiques varient en sens inverse du volume (cf. Fig. 3).

L'augmentation du volume occupé par les équipements se traduit par une diminution des performances du missile : capacité d'emport sur certains véhicules, portée, manœuvrabilité. Ce point doit faire l'objet, dans certains cas difficiles, d'une optimisation au tout début de l'étude.

Le deuxième point est tout à fait spécifique des CCL. L'un de leurs avantages majeurs est de fournir aux boucles de pilotage les vitesses angulaires et accélérations linéaires sous la forme convenable. Le bénéfice est soit d'éviter l'utilisation de boîtiers gyrométriques et accélérométriques spécifiques à cette fonction, soit d'éviter des artifices pour pallier leur absence. Or, les exigences en navigation et pilotage ont des aspects contradictoires qui sont développés un peu plus en détail.

L'exigence principale de la fonction de navigation est la précision. Le choix se porte de manière préférentielle sur les senseurs où l'effet "inertiel" est grand, moment cinétique élevé pour les gyromécaniques, grande surface pour les gyro lasers. La modélisation des effets parasites est très détaillée, les traitements de compensation correspondants sont complexes. La détérioration de la précision induite par l'allongement des temps de traitement est compensée par la meilleure représentation du modèle de fonctionnement du senseur. D'un autre côté, les senseurs doivent avoir une bande passante dont la fréquence de coupure est de l'ordre de grandeur de celle des mouvements du missile induits par son pilotage.

Le pilotage exige des senseurs une bande passante supérieure à celle du missile et des retards de transmission minimaux. Les compensations sont faites pour augmenter la rapidité au détriment des performances. Les solutions technologiques optimales conduisent à choisir des senseurs dont les effets "inertiel" sont plus faibles. Les bruits de mesure sont difficilement filtrables.

Des compromis acceptables ont toujours pu être trouvés. Les mesures inertielles sont fournies sous deux formes : à cadence "lente" avec compensation complète pour la navigation, à cadence rapide et compensation limitée pour le pilotage. Le développement prévisible aujourd'hui de senseurs de toute petite taille, grande dynamique et faible prix rend imaginable la séparation de nouveau des fonctions inertielles de navigation et pilotage, pour les applications les plus contraignantes.

## MOYENS ET PROCÉDURES D'ÉVALUATION DES CCL

L'évaluation des performances d'une CCL est un problème très difficile, particulièrement dans le cas des applications aux missiles tactiques dont les trajectoires sont très mouvementées. Les essais peuvent être séparés en trois classes ayant chacune leurs objectifs propres : les essais en laboratoire, les essais en environnement simulé, les essais en vol réel sur le missile. Les investissements en matériels et logiciels d'essais et d'exploitation sont très importants. La qualité des résultats obtenus en dépend directement.

### LES ESSAIS EN LABORATOIRE

Les moyens d'excitation de la CCL mis en oeuvre à ce stade sont des tables de vitesse angulaire 1, 2 ou 3 axes. Les conditions de travail en laboratoire présentent un avantage majeur : les paramètres d'entrée du système influençant les résultats sont mesurables et facilement reproductibles. Ils sont malheureusement limités en accélération au domaine de la pesanteur. Cette réserve peut être partiellement levée pour certains senseurs dont le comportement est similaire en présence d'accélérations linéaires ou de vitesses angulaires.

Les essais en laboratoire comportent deux phases : les essais "statiques" où les entrées de la CCL sont constantes et les essais "dynamiques" où on recherche au contraire à exciter la CCL dans une large bande de fréquence et en combinant de manière délibérée les mouvements sur les 3 axes du trièdre de mesure. Les essais comportent des études de vieillissement à court terme, de jour à jour, moyen terme (quelques mois), long terme (plusieurs années).

Le diagramme de l'installation d'essai est représenté en figure 4.

Les caractéristiques des tables d'essai sont données dans le tableau 4.

L'exploitation est faite par comparaison des positions angulaires et vitesses linéaires fournies par la CCL aux positions angulaires réelles fournies par la table d'essai (les vitesses linéaires sont nulles). Le traitement, fait en temps différé, fournit par la méthode des moindres carrés les paramètres du modèle d'erreur observables au cours de l'essai. Il exige de grandes précautions en particulier pour l'évaluation des retards apportés par l'installation d'essai. L'utilisation de moyens optiques pour les mesures de position de la table peut résoudre en partie cette difficulté.

Le premier résultat des essais de laboratoire est un modèle expérimental du comportement de la CCL. Il est comparé au modèle théorique. Les différences (il y en a) sont souvent dues à des erreurs d'interprétation des interfaces, mais aussi au fait que certains termes d'erreur ont été négligés. Dans ce dernier cas, il convient de vérifier que les conditions d'essai en laboratoire sont bien contenues à l'intérieur des spécifications d'environnement de la CCL.

Dans le cas particulier des applications aux missiles tactiques manœuvrants, les essais dynamiques combinant des mouvements sur les trois axes sont d'un grand intérêt; ils mettent bien en évidence à la fois les erreurs des senseurs et les erreurs apportées par les algorithmes de navigation.

Le deuxième résultat des essais de laboratoire concerne les procédures et fréquences de calibration de la CCL. Rappelons que cette opération consiste à mesurer ses termes d'erreur sensibles au temps et à mettre à jour périodiquement leur valeur dans sa mémoire. La fréquence de l'opération doit être faible car la calibration exige aujourd'hui dans la grande majorité des cas le démontage de la CCL du missile pour la monter sur une table d'essai. L'observabilité des paramètres à mesurer conduit à donner à la CCL plusieurs positions par rapport à la pesanteur et à la soumettre à des vitesses angulaires largement supérieures à la rotation terrestre. Ce n'est pas le cas pour les plateformes stabilisées.

Le principe des opérations de calibration retenu aujourd'hui est décrit brièvement. Tous les termes sensibles au vieillissement sont observables en essais "statiques" (cf. Tableau n° 1). Le moyen d'essai est une table 2 axes, dont l'axe externe est horizontal, elle permet de placer la CCL dans 8 (cas des performances moyennes) ou 16 positions par rapport à la pesanteur (cf. fig. 5).

L'utilisation d'une table 3 axes peut faire l'économie de 2 positions. Cette solution n'est pas retenue en raison de l'augmentation très significative des coûts de l'installation de maintenance, pour un gain de temps inférieur à 20 %.

Les facteurs d'échelle des gyroscopes sont mesurés par mise en rotation de la table, autour de l'axe interne, dans 3 des positions d'équilibre : la table exécute un ou plusieurs tours suivant la vitesse appliquée et la précision recherchée, pour 2, 4 ou 6 valeurs différentes, de signes opposés 2 à 2.

L'ensemble des opérations dure 1 à 2 h environ, compte tenu des temps d'attente nécessaires à la stabilisation thermique interne des senseurs, garantie de la bonne reproductibilité des conditions de mesure.

Il est envisagé, pour les systèmes de faible précision, de mesurer les facteurs d'échelle des vitesses angulaires au cours de passages de la table d'une position à la suivante.

Enfin, la CCL est introduite dans l'installation de simulation en éléments réels du missile. La table 3 axes qui la supporte exécute les mouvements déduits des ordres envoyés par les chaînes de guidage et de pilotage du missile. L'influence sur le guidage et le pilotage des bruits de mesure, des retards, des interactions et des non linéarités est étudiée. Les précisions de navigation sont partiellement vérifiées.

#### LES ESSAIS EN ENVIRONNEMENT SIMULES

Les performances des CCL et plus particulièrement les précisions et les bruits de mesure sont directement liés aux mouvements du centre de gravité du missile et à ses petits mouvements autour du centre de gravité. Les essais en environnement simulé tentent de reproduire au plus près ces deux types de mouvements en cumulant trois classes d'essais : les essais de qualification, les essais sur centrifugeuse, les essais en vol sur avion d'armes.

Les essais de qualification comportent des essais classiques, vibrations, chocs, accélérations continues, température ; c'est à ce stade qu'est validée la définition de l'équipement. Soulignons l'importance des moyens mis en oeuvre pour simuler l'échauffement cinétique des missiles supersoniques ; ils permettent en effet de simuler très correctement une mission opérationnelle comportant l'emport, la mise en oeuvre, l'initialisation et le vol du missile.

Les essais sur centrifugeuse permettent de combler partiellement les lacunes des essais de laboratoire. Ces machines présentent cependant deux inconvénients : une étroite corrélation entre les vitesses angulaires et les accélérations linéaires appliquées et une très faible dynamique de variation de l'accélération. Il est tout à fait intéressant de compléter les essais par des mesures sur des machines spécifiques, à rayon variable et plateau contrarotatif adapté pour simuler les phases transitoires du vol où le missile subit à la fois de grandes variations d'accélérations et de vitesses angulaires.

Les essais en vol sur avion rapide terminent la phase des essais en environnement simulé. Le programme comprend une dizaine de vols : les premiers sont des vols calmes, puis on exploite progressivement les capacités de manœuvre de l'avion pour rendre observables tour à tour certains termes du modèle d'erreur (ou un groupe de termes).

Il est d'un grand intérêt de disposer à bord de l'avion et tout près de la CCL, d'un système de navigation de grande précision utilisé comme système de référence. Les erreurs de vitesse linéaire et de position sont extraits des mesures de chacun des systèmes en apportant un grand soin aux datations. Un traitement en temps différé permet d'en déduire les principaux termes d'erreur. La complexité du modèle de la CCL laisse cependant une très grande marge d'incertitude sur les autres termes.

La figure 6 présente, à titre d'exemple, les paramètres du modèle excités sur une trajectoire particulière en fonction de la manœuvre en cours.

La figure 7 montre l'évolution des erreurs de vitesses mesurées sur une trajectoire de ce type pour une centrale à gyroscope accordé.

Cette trajectoire, très simple, excite plus de vingt termes du modèle d'erreurs.

#### LES ESSAIS EN VOL REEL DU MISSILE

L'ensemble des travaux réalisés en amont limite l'incertitude qui résulte de la méconnaissance des conditions du vol. La plus grande attention est toutefois apportée à la vérification des ambiances réelles créées par le vol. Les températures et niveaux de vibrations sont mesurés et transmis au sol par l'intermédiaire de la télémétrie ainsi que les informations principales fournies par la CCL. Les méthodes de dépouillement ont été mises au point au cours des essais en vol porté sur avion; on dispose en effet de la trajectographie faite par les moyens optiques ou radars des champs de tir dont on déduit les erreurs de vitesses, élément de base des algorithmes d'identification.

En résumé, nous avons mis en place pour la validation des CCL des moyens et procédures d'essais plus importants que ceux développés pour des systèmes inertIELS des missiles précédents. Il y a deux raisons principales à cet état de fait : premièrement les systèmes d'armes sont plus complexes et les performances, précision, bande passante, étendue de mesure, conditions d'environnement, beaucoup plus contraignantes. Deuxièmement, les CCL présentent des sensibilités à de nombreux facteurs dont certains sont très difficiles à faire apparaître. Le programme présenté suit une démarche progressive et lente, le temps passé en amont est rattrapé par la suite car il subsiste peu d'inconnues au stade des premiers vols du missile.

#### **AVANTAGES ET INCONVÉNIENTS DES DIFFÉRENTES SOLUTIONS POSSIBLES EN REGARD DES OBJECTIFS RECHERCHÉS**

La discussion est ici limitée à deux sujets : la répartition des tâches de calcul entre les équipements et les technologies de gyroscope.

#### REPARTITION DES CHARGES DE CALCUL

Les calculs se rapportant au système inertiel concernent quatre fonctions :

- a) Le traitement des mesures inertielles : bouclage (s'il est digital), filtrage, compensation
- b) Les calculs de navigation : intégration de la vitesse angulaire, calcul de l'attitude, changement de coordonnées des accélérations, calcul des vitesses, des corrections de navigation terrestre, des positions
- c) Les calculs des ordres de guidage et de pilotage
- d) L'initialisation : elle met en jeu des algorithmes spécifiques et aussi, le plus souvent, les calculs a) et b).

L'examen de la répartition de ces calculs entre différents équipements fait apparaître aisément une première conclusion : le traitement des mesures inertielles (a), doit être fait au plus près des senseurs. La première raison est relative à la nature des paramètres à traiter. Les compensations utilisent les mesures des senseurs et de leur température. De plus certaines d'entre elles sont introduites à l'intérieur des boucles d'asservissement. Enfin, les modèles d'erreurs et les grandeurs relatives des paramètres dépendent de la technologie et aussi du type de senseurs d'une même technologie. Le concepteur réalisateur des senseurs est le mieux placé pour optimiser la solution. La deuxième raison est d'ordre technologique. Certains éléments de précision bénéficient ainsi de la relative homogénéité de température qu'il y a dans un équipement.

En ce qui concerne les calculs de navigation fonction (b), la conclusion est moins évidente. L'implanter dans un calculateur spécifique, convient aux applications où les contraintes de précision et de modularité l'emportent sur les contraintes de prix et d'encombrement. La réunion des fonctions mesure inertielles et navigation constitue une CCL autonome, dont la technologie (plateforme ou composants liés) est quasi transparente pour l'utilisateur. On serait alors tenté de réunir les calculs de navigation et les mesures inertielles, mais l'examen de l'implantation de l'initialisation fonction (d), conduit à une conclusion contraire.

L'initialisation met en jeu le missile, son porteur et leurs interfaces. Les procédures et les précisions qui en résultent vont déterminer les temps de réaction du Système d'Armes, les évolutions du porteur avant le tir, donc sa sécurité (cas d'un avion volant à basse altitude). Le Maître d'Oeuvre du Systèmes d'Armes contrôle ces éléments, il est le mieux placé pour définir l'initialisation. Etant devenue distincte des mesures inertielles, l'initialisation est jointe à la navigation dont elle utilise une grande partie.

La séparation des fonctions b) navigation et c) guidage pilotage, va dépendre des applications. Constituer une CCL est avantageux pour les applications haut de gamme où l'on peut sacrifier un peu au confort du technicien : facilité de mise au point des logiciels et transparence de la technologie pour l'utilisateur (nous l'avons déjà vu). Elle présente en outre un avantage particulier pour les applications aux missiles supersoniques. Les équipements étant protégés de l'échauffement cinétique qui se produit au cours du vol, ils n'ont aucun moyen d'évacuer la chaleur qu'il produisent. En limitant le nombre d'équipements en fonctionnement pendant l'initialisation, on diminue leur élévation de température interne : la fiabilité est améliorée.

#### LA TECHNOLOGIE DES GYROSCOPES

Le tableau 5 présente la comparaison de trois technologies de gyroscopes les plus fréquemment utilisées en regard des exigences de nos applications. Il s'agit du gyrolaser, du gyroscope à suspension élastique accordée et des multisenseurs (gyroscopes rendus volontairement sensibles aux accélérations). La quatrième colonne concerne pour mémoire les plateformes stabilisées.

Ce tableau fait apparaître que chacune des technologies envisagées est bien adaptée à une classe particulière d'application.

#### TECHNOLOGIE DES SENSEURS A SUIVRE POUR L'AVENIR

La veille technologique déjà évoquée dans le chapitre traitant des études préliminaires est bien entendu une activité permanente. Son objectif est double : suivre les progrès accomplis par les concepteurs des senseurs et susciter les améliorations des caractéristiques particulières aux missiles tactiques.

Ce chapitre traite brièvement les améliorations que l'on peut attendre des développements en cours vis-à-vis de certains points critiques pour les missiles futurs. Quatre difficultés particulières sont traitées : les étendues de mesure, les bandes passantes et les bruits de mesure, les consommations et les volumes. Les senseurs concernés par cet examen sont les suivants :

- les gyroscopes optiques : gyrolaser actif ou passif et le gyroscope à fibres optiques,
- les gyroscopes à toupie : le gyroscope à suspension élastique accordée, le gyroscope non accordé 2 axes,
- les senseurs utilisant les technologies d'électronique intégrée, les gyromètres à quartz ou les accéléromètres "couches minces" par exemple. Le tableau 6 rassemble les éléments de comparaison.

La situation reste très ouverte : chacune des technologies peut évoluer dans le sens des progrès attendus.

## CONCLUSION

Après plusieurs années d'études et d'essais, nous avons acquis la maîtrise de la technologie CCL appliquée aux missiles tactiques auxquels elle est particulièrement bien adaptée. Les progrès attendus sont liés au progrès des missiles : rapidité, manœuvrabilité, robustesse ; pour leur part, les problèmes liés à la précision sont souvent déjà résolus pour d'autres applications : avions, navires de surfaces ou sous-marins.

Pour nos applications, et en ce qui concerne l'utilisation de la technologie CCL, le pas est franchi ; parmi toutes les solutions envisagées au stade de projet figure toujours une ou plusieurs CCL. Et pourtant, celui qui aborde les CCL pour la première fois peut penser qu'elles sont au delà des capacités des technologies industrielles d'aujourd'hui. En effet, les ordres de grandeur rencontrés à tout propos sont impressionnantes. La liste qui suit en cite quelques uns ; elle est loin d'être exhaustive :

- sophistication de la modélisation des équipements : prise en compte de plusieurs dizaines de paramètres et de phénomènes très rapides ( $10^{-4}$ s)
- puissance de calcul des calculateurs embarqués : 1 Mips/cm<sup>2</sup>
- sensibilité des senseurs :  $10^{-15}$  à  $10^{-18}$  pour les gyroscopes optiques
- perfection des polissages : on serait tentés de changer d'unités et de parler en Angström plutôt qu'en microns !

Et cependant, on est loin d'avoir exploré toutes les capacités de cette technique encore jeune.

Fig. 1 : EXPLOITATION DES MESURES MISSILES

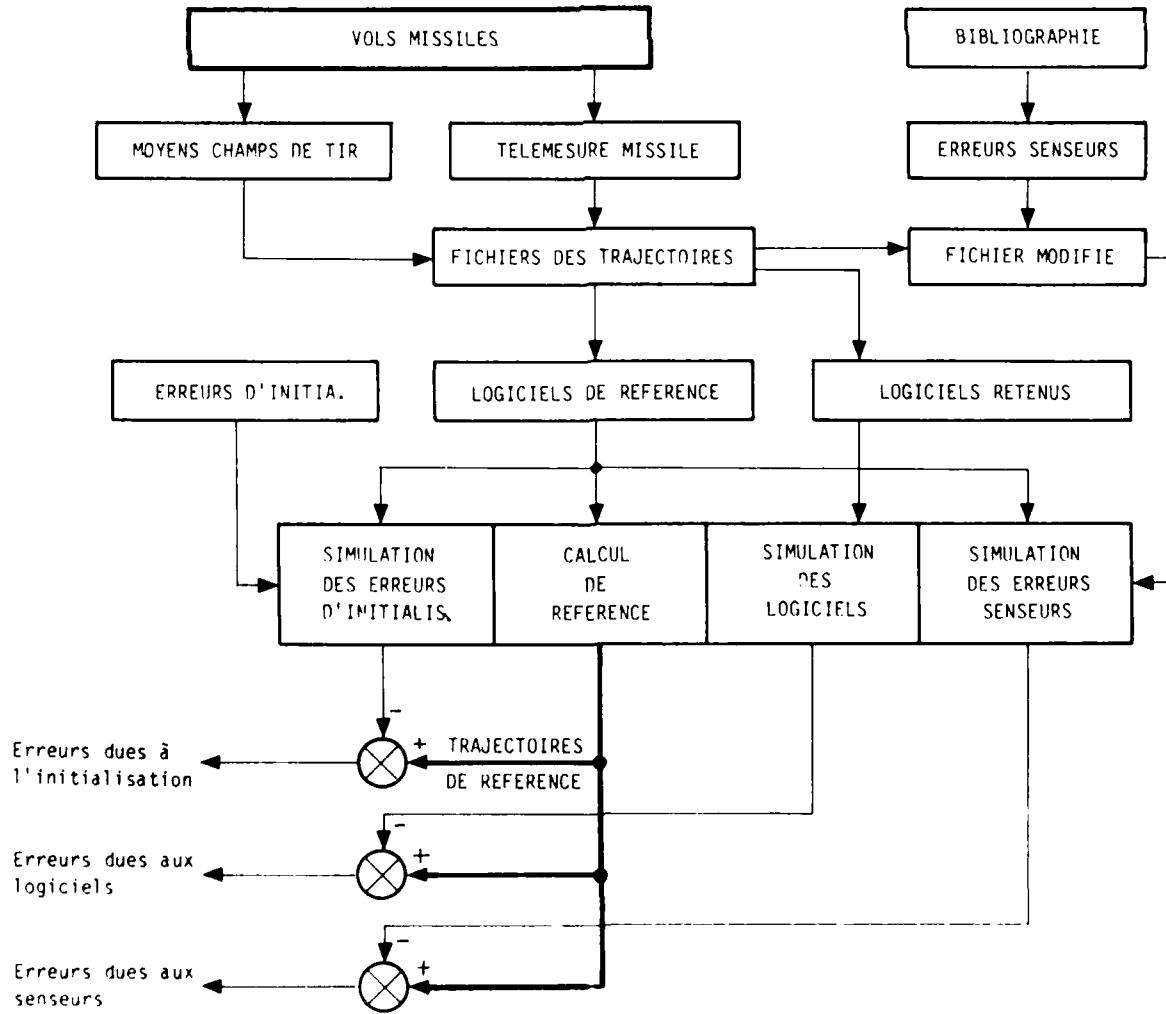


Fig. 2 : ÉCHANGE DES INFORMATIONS

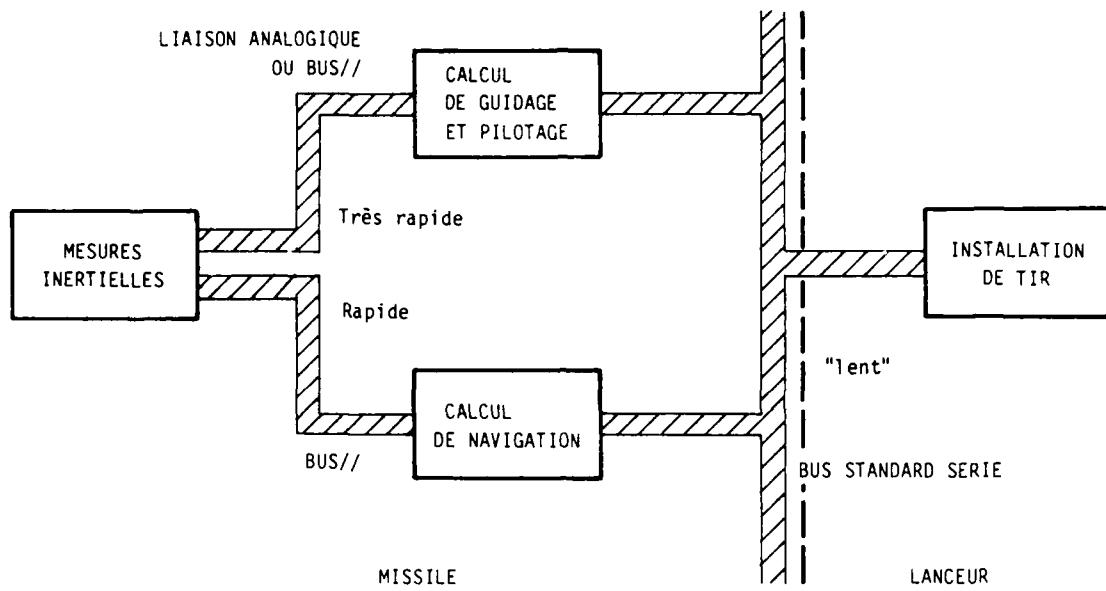


Fig. 3 : VOLUME EN FONCTION DES PERFORMANCES

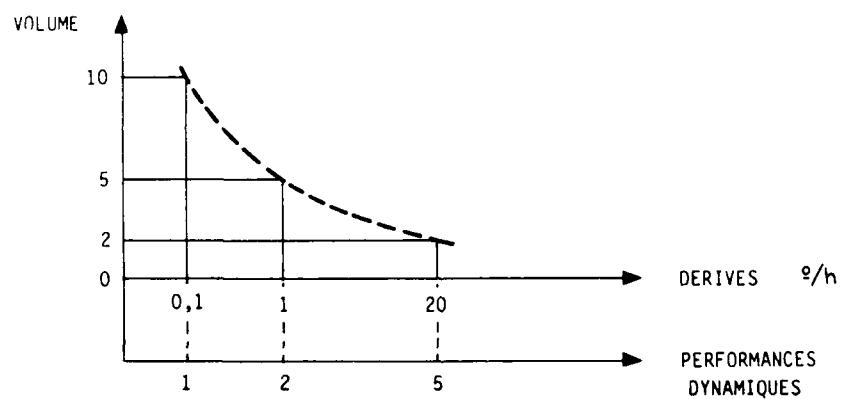


Fig. 4 : BLOC DIAGRAMME DE L'INSTALLATION D'ESSAI

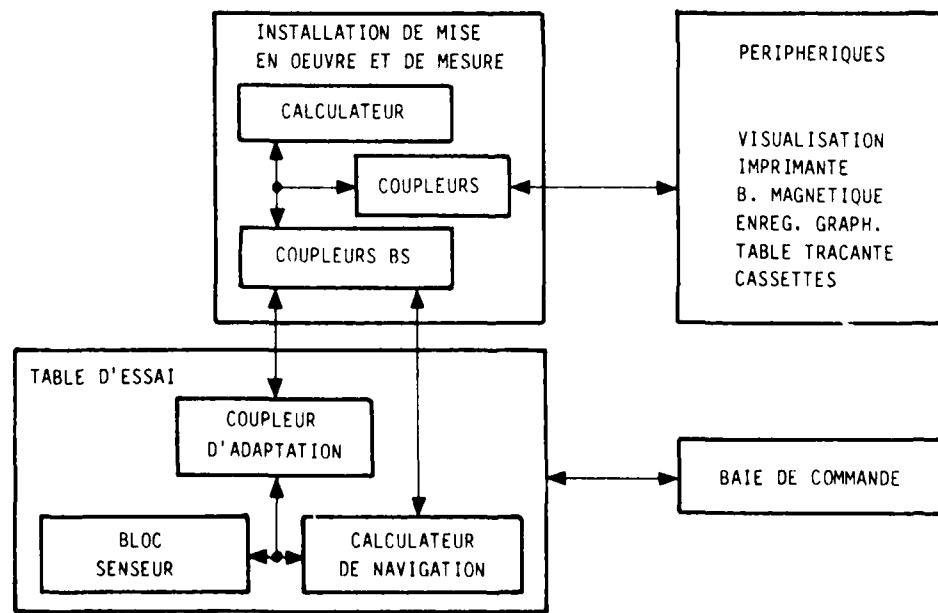
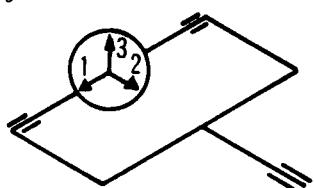


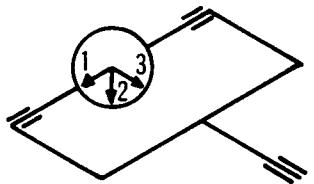
Fig. 5 : POSITIONS DE MESURE EN CALIBRATION

Configuration 1



4 positions par rotation de 90° autour de l'axe externe ou 8 positions par rotation de 45°

Configuration 2 : déduite de la configuration 1 par rotation de 90° autour de l'axe interne



4 (ou 8) positions par rotation de 90° (ou 45°) autour de l'axe externe

Fig. 6 : ERREURS DE NAVIGATION. PARAMÈTRE D'INFLUENCE EN FONCTION DES PHASES DE VOL (EXEMPLE)

Hypothèse : initialisation Vitesse nulle, cap et position par recopie  
Verticale par les accéléromètres de la CCL

INITIAL.DECOLAGE  
ACCELER.MONTEE

① VOL STABILISE

②

VIRAGE DANS LE  
PLAN HORIZONTAL

③ VOL STABILISE

Erreurs ayant une influence :

Phase 1 : Erreur d'initialisation de la verticale + biais  
Erreur accéléromètre vertical

Dérives des gyroscopes (dont 3 termes liés à l'accélération)

Phase 2 : Calage des trièdres au début du virage  
Biais et facteur d'échelle des accéléromètres (au moins 2)  
Facteurs d'échelle gyroscopiques (au moins 2)  
Dérives des gyroscopes liées à l'accélération (différentes des précédentes)

Phase 3 : Erreur de cap initial  
Dérives  
Calage des trièdres en début de phase  
Stabilité des erreurs

Fig. 7 : ÉVOLUTION DES ERREURS DE VITESSE D'UNE CCL EN VOL SUR AVION

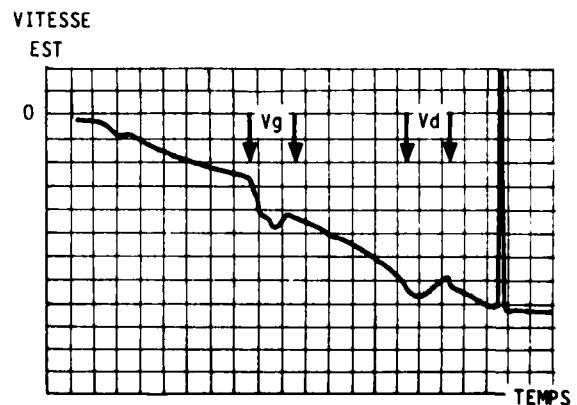
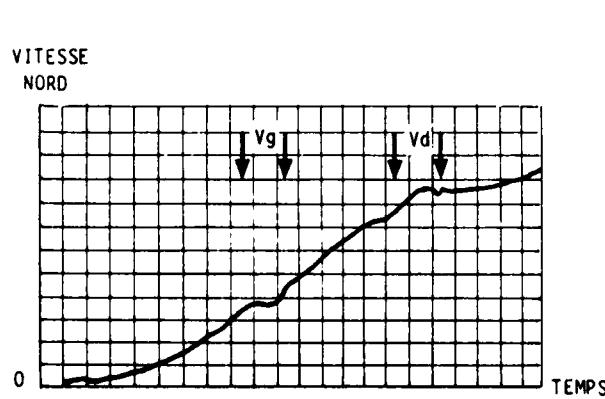


TABLEAU N° 1 : TRAITEMENT DES PARAMÈTRES D'ERREUR

	PRECISION	
	GRANDE	MOYENNE
<b>GYROLASER</b>		
Erreur de zéro	CC	CC
Erreur de facteur d'échelle	CC	C
Calages	CC	CC
<b>GYROACCORDE</b>		
Erreur de zéro	CC	CC
Erreur de facteur d'échelle	CC	CC
Non linéarité du facteur échelle	CC	C
Erreurs liées à l'accélération (balourds au sens large)	CC	CC en partie
Anisoélasticité	C	-
Anisoinertie	C	C parfois
Erreurs dynamiques de la toupie	C	C parfois
Calages	CC	CC
Couplages d'axes	C	-
<b>ACCELEROMETRE ASSERVI</b>		
Biais	CC	CC
Erreur de facteur d'échelle	CC	CC
Non linéarité du facteur d'échelle	CC	-
Anisoinertie	C	-
Couplage d'axe	-	-

Le trait - signifie que le paramètre correspondant n'est pas traité

C signifie que le paramètre est compensé

CC signifie que le paramètre est compensé et calibré.

TABLEAU N° 2 : VOLUME ALLOUÉ AUX SYSTÈMES DE GUIDAGE ET PILOTAGE

	PREMIERE GENERATION	DEUXIEME GENERATION	
		Plate-forme stab.	CCL
Volume du système de guidage et de pilotage rapporté au volume du missile	10 %	5 %	2,5 %

TABLEAU N° 3 : DÉFINITION DES FONCTIONS DE CALCUL

FONCTIONS	ENTREES	SORTIES
MESURES INERTIELLES	Vitesses angulaires Accélérations linéaires Forme physique	Les mêmes informations et/ou les variations d'attitude et de vitesse projetées dans le trièdre missile : ces informations sont compensées Forme analogique et/ou bus
NAVIGATION	Informations inertielles + initialisation par informations de l'installation de tir Bus	Attitudes, vitesses et positions dans le trièdre géographique local Bus
GUIDAGE PILOTAGE	Informations inertielles de navigation de l'installation de tir Bus	Commande des équipements de pilotage Bus ou forme analogique

TABLEAU N° 4 : CARACTÉRISTIQUES DES TABLES D'ESSAIS

FABRICANT	PRINCIPE DE MOTORISATION	DEGRE DE LIBERTE	PRECISION ANGULAIRE	VITESSE ANGULAIRE MAXIMALE	ACCELERATION ANGULAIRE MAXIMALE
GENISCO	Electrique	1	Faible : mesure de la position de la table par moyens optiques	10 rd/s	1 500 rd/s <sup>2</sup>
ACUTRONIC	Electrique	1	50 $\mu$ rd	10 rd/s	150 rd/s <sup>2</sup>
	Electrique	2	25 $\mu$ rd	15 rd/s	100 rd/s <sup>2</sup>
CARCO	Hydraulique	3	"	30 rd/s	1 200 rd/s <sup>2</sup>

TABLEAU N° 5 : PERFORMANCES EN FONCTION DE LA TECHNOLOGIE

EXIGENCES	TECHNOLOGIE				
	GYROLASER (1)	GYROSCOPE ACCORDE (2)	MULTISENSEUR (3)	PLATEFORME (4)	
PERFORMANCES DE NAVIGATION	Elevées	Convient très bien : aussi volumineuse qu'une plateforme	Convient	Ne convient pas	Convient très bien
		Convient bien : plus volumineuse que (2)	Convient très bien	Ne convient pas	Convient : plus chère et plus volumineuse que les trois autres
	Faibles	Pas aujourd'hui : demain le gyro triaxe	Convient	Volume plus faible que (1) et (2)	Ne convient pas
ETENDUE DE MESURE	Grande valeur possible	Compromis avec performances	Idem (2)	Convient très bien	
PERFORMANCES DE PILOTAGE	Bande passante	Elevée au niveau senseur : limitée par le filtrage de l'activation	Peut être très grande avec des artifices dans les boucles	Importante	Ne fournit pas les informations sous la forme convenable
	Bruit	Convient	Convient : compromis avec la bande passante	Idem (2)	
TEMPS DE PREPARATION	≈ 0	Quelques minutes	≈ 0 dans son domaine d'application	> 10 min	
TEMPS DE MISE EN ROUTE	> 2s	Haut de gamme : limité par le warm up	> 2s	> 1 min	
VIEILLISSEMENT	Faible	Plus important que (1)	Pas de vieillissement dans son domaine d'application	Idem (2)	
PARTICULARITES DES MESURES	Allongées par la dérive de marche au hasard	Nombreux éléments d'influence	Test facilité par l'analogie vitesse angul. accéler. linéaire	Très aisées : pas de démontage pour la calibration	
CAPACITE D'INDUSTRIALISATION	Démontrée : reste difficile	Bonne	Bonne	Bonne	

TABLEAU N° 6 : PROGRÈS ATTENDUS

CARACTERISTIQUE	TECHNOLOGIES		
	GYROSCOPES OPTIQUES	GYROSCOPES MECANIQUES A TOUPIE	SENSEURS MECANIQUES INTEGRES
ETENDUE DE MESURE	Très grande capacité, limitée aujourd'hui par les circuits d'interface et de traitement	Compromis avec les performances	Très grandes capacités
BANDE PASSANTE BRUITS	Très grandes capacités (réserve sur le mode de réjection de la zone aveugle)	Très grandes capacités pour les performances moyennes et faibles	Très grandes capacités
CONSOMMATION	Faible	Dépend de la précision : faible en bas de gamme	La plus faible
VOLUME	Les valeurs minimales sont définies par la longueur de la cavité pour les gyroslaser (10 cm) et le diamètre des fibres optiques (quelques cm)	Miniaturisation encore plus poussée (1 cm <sup>3</sup> )	Par principe, le volume est très faible

**STRAPDOWN INERTIAL SYSTEMS  
APPLICATIONS FOR TACTICAL MISSILES  
(STAND-OFF MISSILES)**



Mrs PUECH : ELECTRONIC EQUIPMENT RESPONSIBLE

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92322 CHATILLON CEDEX

STRAPDOWN INERTIAL SYSTEM (SDIS) are particularly well suited for tactical stand-off missile applications. This certainty has been made clear by several years of theoretical design work and trials. Initial theoretical work made wide use of parameters giving detailed definition of missile trajectories and motions as obtained from measurements made on weapons systems then in development or production. Sensors - principally gyroscopes - were laboratory-tested. These preliminary design efforts led to selection of various technologies such as the laser gyro and the mechanical gyro, each being adapted to the weapon system in question. Share-out of functions between various equipment items was defined. Comparison of the advantages and disadvantages of these technologies shows that today the laser gyro is principally retained for high- and medium-range levels of precision, with the mechanical (spin-stabilized) gyro serving the medium- and low precision range. Since requirements are ever increasing in the fields of dynamic performance and robustness, it is here that most progress is expected.

## INTRODUCTION

Until the start of the 'Seventies, inertial navigation systems (most often consisting of stabilized platforms) were reserved for long-range, high-precision complex tactical missiles. The arrival on the market of STRAP-DOWN INERTIAL SYSTEMS (SDIS) was the start of a real technical revolution on prices, volumes and reliability levels, opening up the field of application of inertial systems to cover a wide range of tactical missiles ; stand-off missiles were the first to benefit among our developments.

After several years of design and test work, the results obtained proved to be fully positive : SDIS technology was able to meet all hopes formulated in the fields cited. Moreover, SDIS offered a wide range of equipment items that ensured optimum solutions unavailable via the traditional approaches. It nevertheless remains true that if the obtained solution is more suitable, simpler and cheaper, this is at the cost of increases in theoretical design work and added complexity in test methods - both of which add to development costs.

The present article contains four parts, the first of which examines the field of applications used. The second details preliminary work that establishes the feasibility of the technique for this field of applications, and the third part contains criteria and methods required for establishing selections among the range of technologies available. The fourth part describes the evaluation procedures retained. Finally, after presenting certain elements of comparison between existing systems, the article concludes with a forecast of foreseeable improvements acquired through new technologies still under development.

## FIELD OF APPLICATION

Stand-off missiles vary widely in their assigned rôles, their dimensions and their performances. The parameters influencing the dimensioning of SDIS vary in a more-or-less correlated manner as shown by the following ratios :

Range	1 to 100
Flight time	1 to 100
Speed	1 to 5
Reaction time	1 to 100
Measurement range	1 to 5
Disturbance	1 to 30 (angular and linear vibrations)
Precision	1 to 300 (relatively to target range)
Sensor precision class	1 to 1000
Calculation capacity	1 to 50

Such considerable differences in turn lead to differences (less considerable) in the techniques employed ; nevertheless, design and test methods have a relatively generalized character as will be seen below.

## PRELIMINARY WORK

A technical revolution is always warmly welcomed by the technical fraternity (on account of the new offered possibilities), but also with prudence (on account of the involved risks). Each field presents its own particular difficulties ; inertial systems have a large number of such, one being the impossibility of validating the system outside of its real mode of use. In this instance, the inputs are linear accelerations and angular velocities, which must be known to enable dimensioning of the system, and which must be performed to enable validation. At this project stage, however, the small displacements and vibrations characteristics cannot be known with the required degree of precision (applies particularly to amplitude and phase correlations), and the acceleration profile can only be obtained using a real trajectory. There is therefore a risk that missile flight will lead to wide modification of results obtained by other processes. SDIS are particularly susceptible to such phenomena in view of their close association with missile movement.

The design work aimed at establishing the feasibility of SDIS technology concentrated on sensors (essentially gyroscopes), processing algorithms and computers.

### THEORETICAL WORKS

The purpose here was two-fold : to define approximately the performance of the SDIS for the envisaged applications, and to define navigation software.

The first step was to produce a working tool enabling evaluation of the solutions considered, to highlight the advantages and disadvantages. This consists of gathering data on the inertial system inputs, that is, defining the missile movements characteristics : these include parameters associated with the missile trajectory and small motions around the trajectory.

This work relied largely on the results of measurements performed on missiles currently under development or in production. The trajectory was characterized by test-site radars and optical instrumentation ; missile attitude, accelerations and vibrations were computed from measurements obtained within the missile and transmitted by data link. The results obtained centered on :

- Several characteristic trajectories defined by accelerations and angular velocities sampled at a very high rate,
- spectral distribution of linear accelerations and angular velocities as per the various phases of flight.

This tool was employed in three projects : evaluation of navigation errors deduced from theoretical modelling of sensor behaviour, evaluation of the influence of initializing errors, and definition of the navigation software. The organizational method is illustrated in Figure 1.

The first task consisted of defining a reference trajectory, deduced from the files by a highly accurate calculation method (rapid dual-precision Runge-Kutta integration to an order of 4).

Sensor errors were defined by bibliographical study work and contact with the LRBA (Laboratoire de Recherche Balistique et Aérodynamique) and sensor manufacturers. These errors were introduced into the files defining the trajectories. Navigation errors were obtained by comparison with the reference trajectory. This project enabled compilation of sensor technical specifications initial version.

Study work into initialization errors requires no comment.

The software design stage concentrated on the following points : rotation operator choice, order of angular velocities integration, computing rates, LSB value and mathematical operations precision level.

The rotation operator was selected independently of the trajectories required. As with other users, the method of quaternions was preferred to that of director cosines or Euler angles. The quaternion is in fact well adapted to real-time mathematical calculation since it does not rely on trigonometric use quantities, and retains the trihedrals orthogonality. Additionally, prohibited positions do not exist.

Other elements were obtained using simulated trajectories. Overall results were as follows :

- The integration order is generally equal to 3 ; it can be higher for certain applications where variations in the inputs size is very fast. For identical performance levels, the solution retained was that producing the most complex algorithm for the slowest rate.
- Computations are performed at two rates in a ratio of 10 (approximately). The attitude and speed computations are performed at the higher rate, with the position, earth navigation correction and quaternion normalizing computations being performed at the lower rate.
- Quantification of data required for control is subject to greater degrees of constraint than data required for navigation : this field is therefore covered elsewhere.

Algorithms are furthermore verified in the presence of vibrations and rotations with an average nul value (conical motions in particular), to evaluate robustness in the face of such perturbations.

#### SENSOR RESEARCH

Promising technologies can be identified several years before their appearance on the market by permanent technological monitoring. Since accelerometers were already available, the main monitoring effort was towards gyroscopes. The performance requirements were unusual for tactical missile equipment applications : the sensor dynamics (in fact the measurement range divided by the zero error), is  $10^5$  for medium performances, and reaches  $10^8$  for high performance levels.

Gyroscope manufacturers made available equipment reputed able of such performances. It was then necessary to perfect the required laboratory equipment, materials, procedures and processes to allow verification. This work was aimed at both mechanical and laser gyros. Error models adapted to the tactical missile application were refined and applied to the trajectories obtained during the theoretical study phase. The detailed technical specifications of the sensors were then deduced for each considered application.

Table 1 shows the main parameters purposed to requirements, and the type of processing retained : parameters may or may not be processed or compensated. "Compensation" consists of deducing from the equation models, the influence of the used parameters (and whose values were previously stored in a memory). Certain of these values vary with time and require periodic updating - this is referred to as "calibration". The table has been designed for the three most commonly used types of sensors - the laser gyro, the drytuned gyroscope and the pendular accelerometer - and for two classes of application : high - and medium-precision.

COMPUTERS

Computers were for a long time the main obstacle to SDIS technology. Considerable progress achieved in the 'Seventies enabled removal of this barrier. The design efforts concentrating on computers structure, basic components and languages.

Preliminary work established the feasibility of SDIS technology for the main part of tactical missiles, including stand-off missiles, except (temporary) for the most accurate missiles. This phase also highlighted the most stringent constraints, which would imperatively require integration in the later guidance and control studies.

**CRITERIA AND METHODS OF SELECTION**

Once the feasibility of SDIS technology was established, applications were immediate. Selections were required from among the considered solutions. The first part of this section presents the retained criteria, followed by an outline of the general characteristics of the selected solutions, and concluding with examination of two points presenting a difficult compromise.

CRITERIA

The selection criteria did not differ fundamentally from those of any electronic equipment. The relative weight of any such criteria depends on the required use.

The classification presented here is of a representative nature only, given the diversity of stand-off missile types.

The first criterion concerns volume : despite considerable inroads into this field, volume remains a constraint of the first degree : this is shown significantly by comparison of first- and second-generation anti-ship missiles (Table 2).

(It should be noted that present-day ranges are three to four times greater).

Although the control and guidance system occupies a minimum volume when compared to the overall missile volume, it remains nevertheless advantageous to achieve even greater reductions. When it is realized that, for example, a 1-centimetre saving in length represents a 1-kilometre range increase.

A second criterion of capital importance concerns constraints on the control and guidance system resulting from the missile on its launch platform : whether this is a land vehicle, helicopter, ship or aircraft, it must conserve maximum operating capability during the readying phase in which the inertial system is initialized. Criteria to be taken into account include duration of readying phase (preheating), duration of initializing phase, carrier platform maneuvering limits, or, conversely, imposed trajectories.

The third criterion concerns the capability of the equipment to withstand the operational storage and utilization conditions of the weapon system. Even after several years of storage, the system must be operational without any destorage process other than operational use. Preventive maintenance operations must remain few, simple and fast. The optimum situation allows use of tactical missiles as ordinary ammunition rounds.

Costs are also a fundamental element of choice. As with volume (although to a lesser extent), the costs of inertial systems have decreased considerably in recent years, with SDIS technology itself playing an important rôle in this trend. Additionally, the growing complexity of weapon systems further reduces the cost share of inertial systems. Nevertheless, SDIS remain a significant cost factor, particularly during development.

Performance appears to occupy a poor position in this criteria breakdown, because of the fact that the technologies from which the selections were made are already supposed to have met minimal requirements. The performance criterion thus plays a hit-and-miss rôle : if levels are not met, the system is rejected. Where performance exceeds the minimal specifications, it remains a beneficial (but not overriding) factor by offering wider envelopes for constraints imposed on other functions.

The last criterion examined here concerns the adaptive capabilities of the technologies retained. Mention was made above of development costs importance : these will be more rapidly redeemed if a technology studied and validated elsewhere can be used in a different application.

### RETAINED SOLUTIONS GENERAL CHARACTERISTICS

Despite the wide dispersion of tactical stand-off missile applications and of the importance attributed to the principal criteria in each case, the retained solutions follow certain general principles. These are briefly examined below.

#### GENERAL STRUCTURE

The most characteristic choice element concerns the more-or-less high concentration of missile computing functions ; they are represented in Table n° 3, limited to functions using inertial measurements. As far as possible, the centralized structure is selected. Nevertheless computations can to a greater or lesser extent be assembled into certain applications where volume or cost constraints may be harder to respect than actual performance. This is discussed in detail below.

#### Data transfers

It is rarely possible to use a single transmission mode. The fastest data (essentially control signals) are delivered in analog form or via a high-speed parallel bus serving a limited number of users. Slower transfers use a standard serial bus to allow interfacing (where possible) with similar buses used on launch equipment. This is illustrated in Figure 2.

#### Sensors

The question of employing a single technology does not arise : each available technology corresponds to a certain performance range with overlay areas where, for example, gyroscopes as widely dissimilar as laser gyros and drytuned gyros find themselves competing. Within the budgetary capacities of a given programme, it is often possible to install two different interchangeable technologies to meet the weapon system requirements in almost identical fashion. Nevertheless, it is to be noted that certain characteristics such as noise or dynamic performance are specific to sensor types, and the computer must be able to recognize which sensor is being used, to allow selection of the appropriate software.

#### Computers

As with sensors, there is no question of defining a unified structure or technology. Mention can nevertheless be made of the advantages of microprogrammable computers for repetitive real-time computations, data transfers and specific mathematical operations etc.

### SELECTION COMPROMISES

Selection criteria are numerous, and are therefore never fully satisfied. This section examines two special compromise situations encountered systematically by SDIS applications in stand-off missiles. They concern contradictions between performance-volume requirements and between data required for navigation and guidance.

It is obvious that inertial system precision increases with its volume. It is more arguable but generally true that dynamic performance varies inversely with volume - see Figure 3.

Increases in the volume occupied by the equipment translate directly into reduced missile performance, variously affecting payloads, ranges or maneuvering capability. In certain difficult cases, therefore, this point requires optimizing at the very start of the design stage.

The second point is totally specific to SDIS systems. One of the main advantages of such systems is their ability to provide the guidance loops with angular velocities and linear accelerations in suitable form - the benefit being to avoid the need for rate gyro and accelerometers units specific to these functions, or to dispense with elaborate alternatives. However, navigation and guidance functions requirements present contradictory aspects as examined below.

Precision is main requirement of the navigation function. System selection tends to prefer sensors getting a large "inertial" effect, that is, high kinetic moment for mechanical gyros or large surface for laser gyros. Parasite effects modelling is performed in detail, the corresponding compensatory computations being complex. Precision losses resulting from increases in processing time are compensated for by sensor operating model optimized presentation. From another angle, sensors bandwidth are in the same order as missile motions induced by the control system.

The control function requires sensors with pass bands higher than that of the missile proper, and minimal transmission delay. Compensations are performed to increase rapidity at the expense of performances. The optimum technological solutions tend towards sensors with lower "inertial" effects. Measurement noise presents filtering problems.

Acceptable compromises have always been found. The inertial measurement unit delivers measurement signals at two rates - "slow" with full compensation, for navigation, and "fast" with limited compensation for control. Development trends seen today suggest very small sensors with high dynamic capabilities and low costs, leading to possible separation once more of the inertial navigation and control functions, in the most stringent applications.

## SDIS EVALUATION MEANS AND PROCEDURES

SDIS performance evaluation is a complicated problem, particularly in the case of tactical missiles where the trajectory is complex. Trials can be separated into three classes, each with its specific aim, as follows : laboratory testing, simulated environment testing and flight tests on the actual missile. Investment in test and operating equipment and software is very high, and directly influences the quality of the results obtained.

### LABORATORY TESTS

SDIS test equipment at this stage are 1, 2 or 3-axis angular velocity tables. The laboratory offers major advantages for this work - the input parameters of the system influencing the results can be measured and easily reproduced. They are unfortunately limited to gravity as far as accelerations are concerned. This reserve can be cancelled partially for certain sensors whose performance is similar under linear accelerations or angular velocities.

Laboratory testing consists of two phases - "static" tests, where the SDIS inputs are constant, and "dynamic" tests, where, on the contrary, it is attempted to simulate the SDIS over a wide frequency band while combining planned movements along the three axes of the measurement frame. Testing also contains short-term ageing tests, daily, medium-term (monthly) and long-term tests (several years).

The test set-up diagram is shown in Figure 4.

The test table characteristics are given in Table 4.

The test method consists of comparing the angular positions and linear velocities delivered by the SDIS, with the actual angular positions recorded by the test table (linear velocities are null). Processing is performed off line and uses the least-squares method to supply error model parameters as observed during the test. This demands stringent precautions, particularly as regards evaluation of delays caused by the test installation. Optical means use for measurement of table position can in part resolve this problem.

The first result from the laboratory tests is an experimental model reproducing the behaviour of the SDIS. This is then compared with the theoretical model. Any differences may be due to interpretation errors at the interfaces, or to the fact that certain error terms were omitted. In the latter case, it is necessary to check that the laboratory test conditions are within the environmental SDIS specifications.

In the special case of agile tactical missiles, dynamic tests combining movements in all three axes are of special interest, since they highlight both sensor errors and errors induced by the navigation algorithms.

The second result from the laboratory tests concerns SDIS calibration procedures and frequencies. It should be noted that this operation consists of measuring time sensitive error terms and of periodic updating of the corresponding values in the memory. The frequency of this operation should be low, since present-day calibration procedures in most cases involve removal of the SDIS from the missile for installation on a test table. Indeed parameters measurement consists of setting the SDIS to several gravity positions and then submitting it to angular velocities far greater than the earth's rotational value. This is not required for stabilized platforms.

A brief outline of present-day calibration operations is contained below. All terms susceptible to ageing are observable in "static" tests (see table 1). The test equipment consist of a two-axis table, with the external axis horizontal. This enables setting the SDIS to 8 or 16 positions as defined by gravity (for medium and normal performance requirements respectively). See Figure 5.

Use of a three-axis table can save two positions. This solution is not retained in view of the significantly increased installation and maintenance costs that allow a time saving of less than 20 %.

The gyroscope scale factors are measured by spinning up the table around its internal axis, in three stable positions : the table executes one or more lap, depending on the speed input and the precision required, for 2, 4 or 6 different values, in opposed pairs of signs.

This operation series lasts approximately one to two hours, allowing for the waiting periods required for internal thermal stabilizing of sensors (which is the guarantee of reliable reproduction of measurement conditions).

For low-precision systems, it is envisaged to measure the angular velocity scale factors during passage from one table position to another.

Finally, the SDIS is inserted in a simulator that consists of actual missile parts. This is mounted on a three-axis table that performs movements deduced from command signals transmitted by the missile control and guidance channels. The influence of measurement noise, delays, interactions and non-linearity on the control and guidance function is studied. Navigation precision is partially verified.

#### SIMULATED ENVIRONMENT TESTS

SDIS performance, and more especially measurement performance and noise, are directly associated with around the missile's centre of gravity motions and small-scale displacements about this point. Simulated environment tests attempt to reproduce as exactly as possible these two types of movement by cumulation of three types of tests - qualification tests, centrifugal tests and in-flight testing on military aircraft.

Qualification tests consist of the traditional tests - vibration, drop-tests, continuous acceleration and temperature. At this stage the equipment definition is validated. It is worth to notice the sophisticated nature of the means employed for simulation of the kinetic heating of supersonic missiles ; an operational mission can be very accurately simulated to reproduce the captive flight phase, readying, initializing and actual missile flight. Centrifuge testing enables solving of some of the problems left unanswered by the laboratory tests. Centrifuge machines present, however, two disadvantages - a strict correlation between angular velocities and the linear accelerations applied, and a very low dynamic rate as regards variations in acceleration. It is of the utmost interest to complete testing on these machines by measurements on specialized machines featuring variable radius control and contra-rotating tables adapted to simulate transitory flight modes where the missile is simultaneously subject to large variations of acceleration and angular velocities.

In flight testing on high-speed aircraft represents the end of the simulated environment tests. The programme consists of approximately ten flights, the firsts having a passive nature. These are followed by progressive exploration of the aircraft's maneuver range to enable observation of successive error model terms (or groups of terms).

It is an advantage to have a high-precision navigation system aboard the aircraft, in close proximity to the SDIS, for use as a reference system. Errors in linear velocities and position are extracted by measurements from each system, with close attention paid to dating. Off-line processing then allows the principal error terms inférence. Nevertheless, the SDIS model complexity allows for a wide range of incertitude as regards other terms.

Figure 6 is an example of model parameters energized for a given trajectory associated with a given maneuver.

Figure 7 shows the error evolution for velocities measured on such a trajectory, with a dry tuned gyroscope SDIS. It may be noted that such a simple trajectory involves implementation of more than twenty terms in the error model.

#### MISSILE FLIGHT TESTS

The preceding study work limits the degree of uncertainty resulting from the untested nature of the actual flight conditions. Nevertheless, the greatest attention is paid to actual flight ambiances verification : temperatures and vibration levels are measured and transmitted to the ground by a data link, along with principal data transmitted by the SDIS. Processing methods were already developed during the captive flight program. Trajectory plotting information is obtained by optical means or radars at the firing site ; using this data, it is possible to deduce the velocity errors that represent the basic element of the identification algorithms.

To sum up the means installed for validation of SDIS technology consist of greater test resources and procedures than was the case for the inertial systems of previous missile generations. This situation results from two main causes-firstly, the present complexity of weapon systems demands higher levels of precision, wider band width, wider measurement ranges and harsher environmental criteria.

Secondly, SDIS present increased sensitivity to numerous factors, certain of which are extractable only with great difficulty. The program presented herein follows a progressive and often slow development : however, time spent in earlier stages of the program is made up for subsequently, since few unknowns remain at the missile test phase onset.

## ADVANTAGES AND DISADVANTAGES OF THE VARIOUS SOLUTIONS AVAILABLE

The following discussion is limited to two subjects : share-out of computing tasks between various equipments, and gyroscope technology.

### SHARE-OUT OF COMPUTING TASKS

Computing tasks associated with the inertial system concern four functions, as follows :

- a) Processing of inertial measurements : slaving (if not digital), filtering and compensation.
- b) Navigation computing, including angular velocity integration, attitude calculations, acceleration coordinates change, velocities calculation, earth navigation corrections and position.
- c) Control and guidance computation.
- d) Initializing : this entails specific algorithms and, in most cases, functions a) and b) above.

Examination of share-out of these functions between the various equipments immediately allows a first conclusion to be reached : processing of inertial measurement (a) must be performed as near to the sensors as possible. The first reason is the nature of the parameters for processing : compensation systems use the sensor measurements and temperature. Additionally, certain compensations are performed inside the slaving loops themselves. Finally, error models and the parameters relative scales of size depend on the technology used, and also for some technology on the sensors of a given type. The constructor who produces his own sensors is best placed to find the optimal solution. The second reason is of a technological nature : certain high-precision components benefit from the relative degree of temperature stability afforded by an equipment. This structural approach has a clear interface, in that the IMU delivers directly usable data.

As regards, the navigation processing function (b), the conclusion is less obvious. Implantation of this function into a specific computer - the navigation computer - is a satisfactory solution for applications where precision constraints and modular layout are more important than cost or volume. Integration of inertial measurement and navigation computing forms an autonomous INS whose technology (platform or strapdown) is practically "transparent" from the user point of view. It is therefore a temptation to incorporate navigation computations into the IMU, but the functional set-up of (d) - initializing - examination leads to the opposite conclusion.

Initializing brings the missile, its carrier and the respective interfaces into play. The procedures and precision levels attained at this stage will influence the reaction time of the weapon system, the maneuver range of the carrier platform prior to launch, and thus safety (as illustrated by the case of an aircraft flying at low altitude). The weapon system Primary Contractor has control of these elements and is best placed to define the initializing function. Now distinct from the IMU, this function is associated with the navigation function, on which it relies to a great extent.

Separation of functions (b) - navigation, and (c) - control and guidance - depends on particular applications. Building of an SDIS is advantageous for top-range applications, where sacrifices can be made to assist the technician (such as easier software design), or for technology transparency for the user (as seen above). It additionally offers a particular to supersonic missiles advantage : since equipments are protected from the kinetic heating generated by flight, they have no means of evacuating the heat they themselves produce. Thus by limiting the number of equipments operating during the initializing phase, internal temperature increases are kept to a minimum, thus increasing reliability.

## GYROSCOPE TECHNOLOGY

Table 5 contains a comparison of the three most commonly used gyroscope technologies, from the viewpoint of the application requirement contained herein. The gyroscopes examined are the laser gyro, the dry tuned gyro, and multisensors (gyroscopes made sensitive to accelerations). The fourth column contains details of stabilized platforms, for reference.

It appears, each envisaged technologies is well suited at each particular application.

## TECHNOLOGY OF SENSORS FOR FUTURE DEVELOPMENT

The technological monitoring referred to earlier in the "Preliminary Work" section is naturally an ongoing activity. Its purpose is two-fold : to follow the progress achieved by sensor designers, and generate improvements in the particular characteristics of tactical missiles.

This section briefly examines the improvements expected from present developments as regards certain critical points affecting future generations of missiles. Four special difficulties are examined : measurement range, band width and measurement noise, consumptions, and volumes. The sensors concerned herein are the following :

- optical gyroscopes - active or passive laser gyro, and optical-fibre gyroscope,
- wheel gyroscopes - dry tuned gyroscope and twin-axis untuned gyroscope,
- sensors using integrated electronics, such as quartz rate gyros, integrated accelerometers.

Table 6 presents any comparison elements.

It appears, numerous difficulties will be soon solved.

## CONCLUSION

After several years of research and testing efforts, SDIS technology for tactical missile applications has now been acquired, and shows itself as particularly suited to such applications. Expected improvements are centered on progress in missile technology proper - speeds, manoeuvrability and robustness. As regards problems of precision, these have in many cases already been resolved for other applications such as aircraft, surface ships or submarines.

As regards the present application, and from the SDIS technology point of view, the first steps have already been taken. Among all the considered solutions for the predevelopment phase, one or more SIDS appear. Nevertheless, the scientist which has to deal with them for the first time, may think they are beyond today industrial capacities. Indeed, the encountered scale orders even turn are impressive. The following list mentions some, it is far to be exhaustive :

- sophistication of equipment modelling, including processing of scores of parameters and rapid phenomena ( $10^{-4}$  secs),
- calculation power of on-board computers of 1 Mips/cm<sup>2</sup>,
- sensor sensitivity levels of between  $10^{-15}$  and  $10^{-18}$  for optical gyroscopes,
- mirror-polishing levels where the measurement units might more appropriately be the Angström rather than the micron.

Nevertheless, the full possibilities of this new technology are far from fully explored.

Fig. 1: PROCESSING OF MISSILE MEASUREMENTS

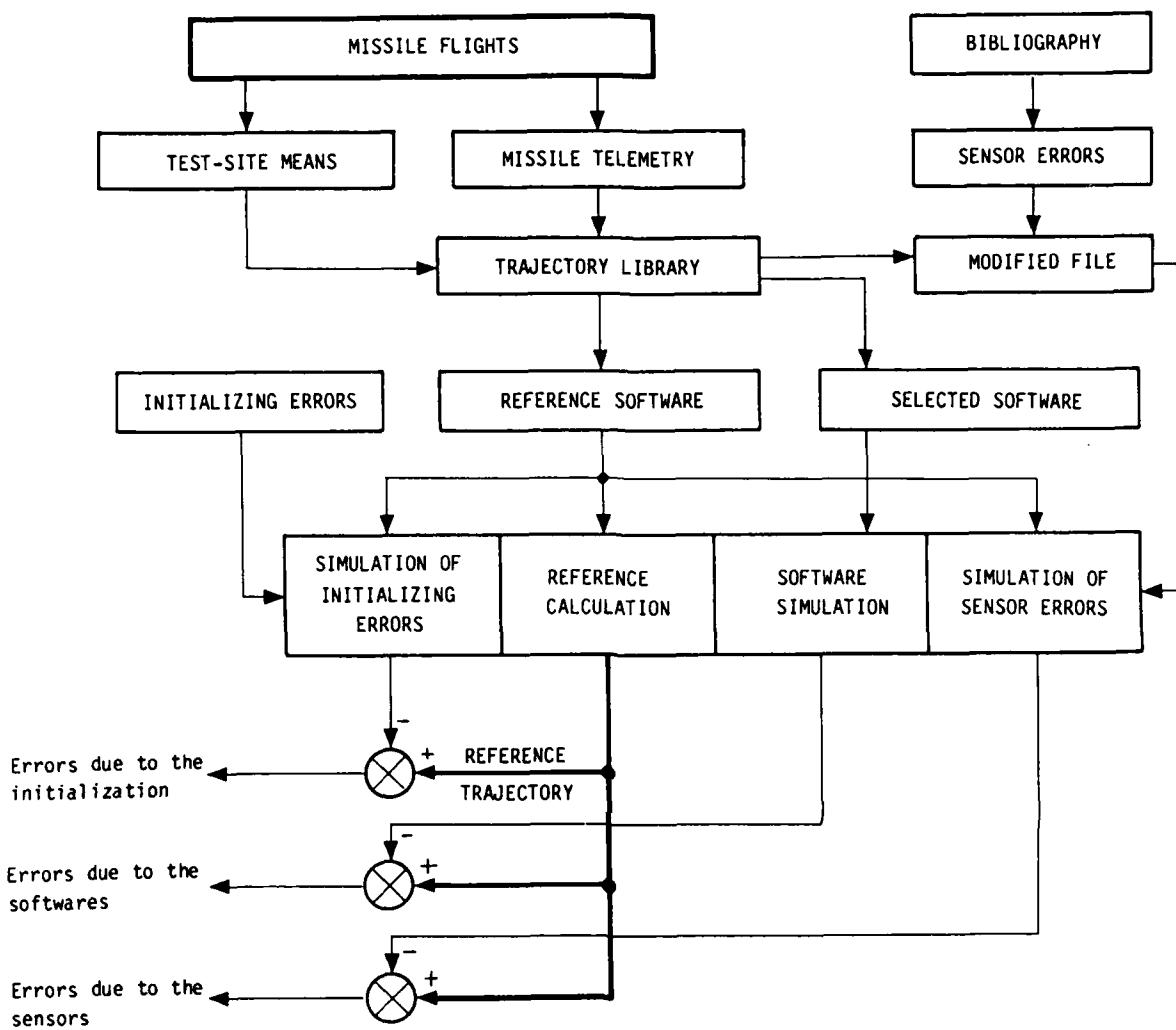


Fig. 2: DATA TRANSFERS

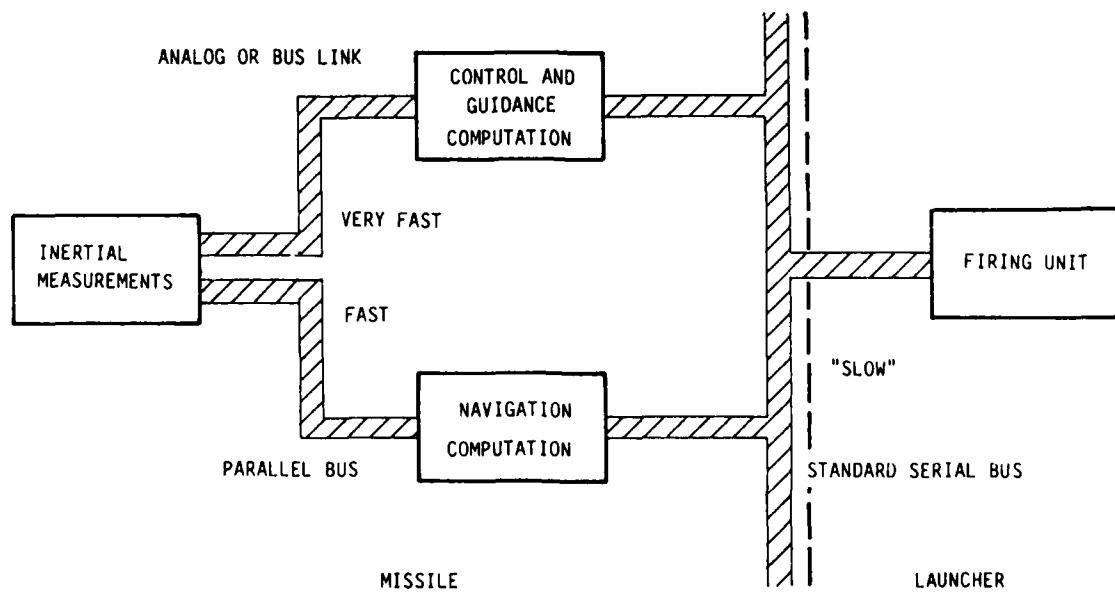


Fig. 3: VOLUME VERSUS PERFORMANCE

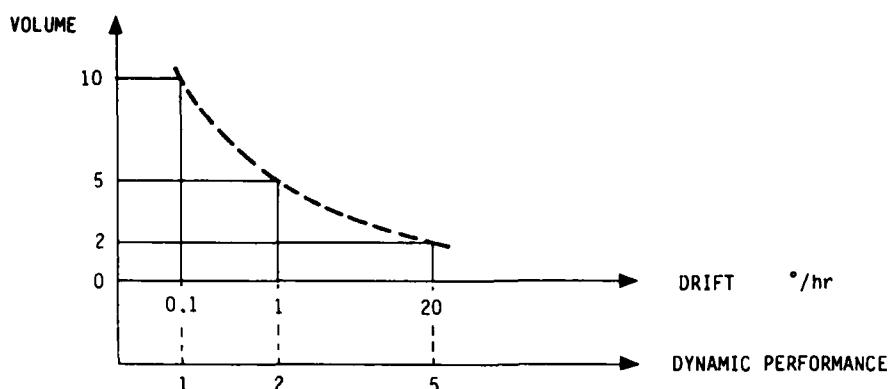


Fig. 4: TEST SET-UP BLOCK DIAGRAM

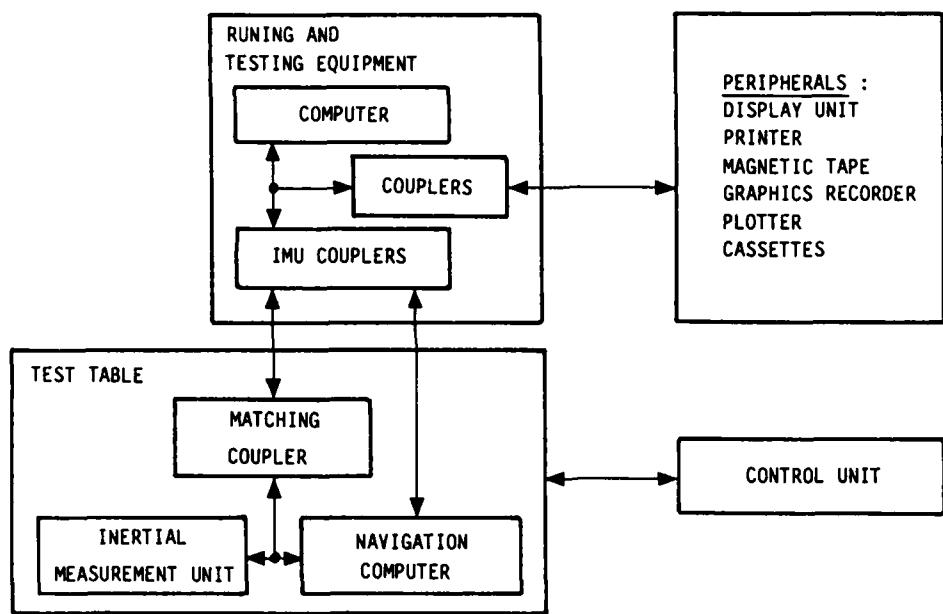
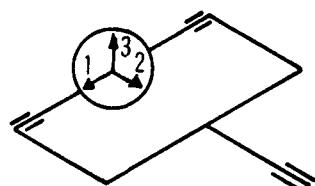


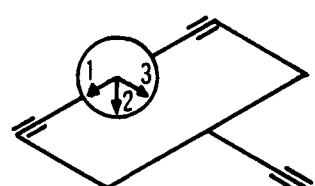
Fig. 5: MEASUREMENT POSITIONS FOR CALIBRATION

configuration 1



4 positions by rotating 90° around the external axis, or 8 positions by 45° rotation

Configuration 2 (deduced from configuration 1 by rotation 90° around the internal axis)



4 (or 8) positions by rotating 90° (or 45°) around the external axis

Fig. 6: NAVIGATION/INFLUENCE PARAMETER ERRORS FOR VARIOUS PHASES OF FLIGHT

Hypothesis : Initializing

Velocity : Null - Heading and position by copy vertical by SDIS  
accelerometers slavingINITIALIZING. TAKE-OFF.  
ACCELERATION. CLIMB-OUT

1 STABILIZED FLIGHT

2

HORIZONTAL TURN

Observable errors

3 STABILIZED FLIGHT

Phase 1 Vertical initializing error

Gyroscope drift (thus, three terms associated with acceleration)

Vertical accelerometer error

PHASE 2 Mispositioning of the frame at start of turn

Gyroscope scale factors (at least two)

Gyroscope drift associated with acceleration (differs from previous errors)

Accelerometer bias and scale factor

PHASE 3 Errors stability

Drift

Mispositioning of the frame at start of phase.

Fig. 7: VELOCITY ERROR CURVE FOR SDIS INSTALLED ON AIRCRAFT

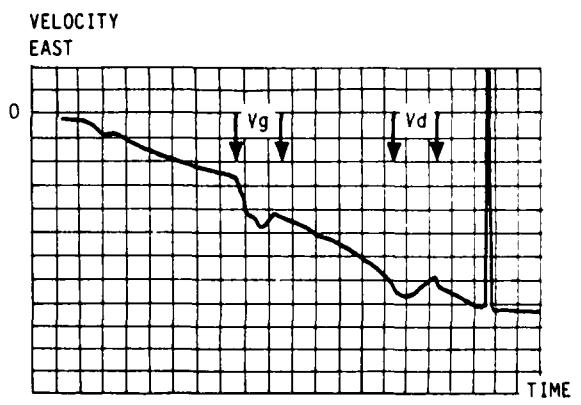
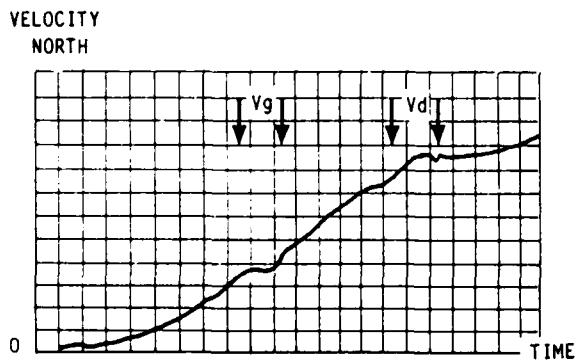


TABLE 1 - PROCESSING OF ERROR PARAMETERS

	PRECISION	
	GRANDE	MOYENNE
LASER GYRO		
zero error	CC	CC
scale factor error	CC	C
positioning	CC	CC
DRY TUNED GYRO		
zero error	CC	CC
scale factor error	CC	CC
scale factor non-linearity	CC	C
errors depending on accelerations (all types imbalances)	CC	CC (partly)
anisoelasticity	C	-
anisoinertia	C	C (some-times)
rotor dynamic error	C	C (some-times)
positioning	CC	CC
cross axis error	C	-
SLAVED ACCELEROMETER		
bias	CC	CC
scale factor error	CC	CC
scale factor non-linearity	CC	-
anosoinertia	C	-
cross axis coupling	-	-

A dash (-) indicates that the corresponding parameter is not processed.

"C" indicates that the parameter is compensated.

"CC" indicates that the parameter is compensated and calibrated.

TABLE 2 - VOLUME ALLOTTED TO CONTROL AND GUIDANCE SYSTEMS

FIRST GENERATION	SECOND GENERATION	
	Stabilized platform	SDIS
Volume of control and guidance package as against overall missile volume	10 %	5 %
		2.5 %

TABLE 3 - DEFINITION OF COMPUTATION FUNCTIONS

FUNCTION	INPUTS	OUTPUTS
INERTIAL MEASUREMENTS	Angular velocities Linear accelerations Physical form	Same data and/or attitude/speed variations projected in the missile frame. This data are compensated Analog and/or bus form
NAVIGATION	Inertial measurements + initializing by data delivered by launch installation Bus	Attitudes, velocities and positions in the local geographic frame Bus
CONTROL AND GUIDANCE	Data delivered by the IMU, the NAVIGATION and launch installation Bus	Control of guidance equipment Bus or analog form

TABLE 4 - CHARACTERISTICS OF TEST TABLES

MANUFACTURER	MOTORIZATION	DEGREES OF FREEDOM	ANGULAR PRECISION	MAX. ANGULAR VELOCITY	MAX. ANGULAR ACCELERATION
GENISCO	Electric	1	Low (table position measured by optical means)	10 rd/s	1 500 rd/s <sup>2</sup>
ACUTRONIC	Electric	1	50 $\mu$ rd	10 rd/s	150 rd/s <sup>2</sup>
	Electric	2	25 $\mu$ rd	15 rd/s	100 rd/s <sup>2</sup>
CARCO	Hydraulic	3	"	30 rd/s	1 200 rd/s <sup>2</sup>

TABLE 5 - PERFORMANCE VERSUS VARIOUS TECHNOLOGIES

REQUIREMENT	TECHNOLOGY			
	LASER GYRO (1)	DRY TUNED GYRO (2)	MULTISENSOR (3)	PLATFORM (4)
NAVIGATION PERFORMANCE	Very suitable : same volume as a platform	Suitable	Unsuitable	Very suitable
MEASUREMENT ENVELOPE	High value possible	Compromise with performance	Same as (2)	Very suitable
GUIDANCE PERFORMANCE LEVEL	High for sensor ; limited by locking zone cancellation	Can be very high if loops are specially configured	Very high	Does not deliver data of the suitable size
PREPARATION TIME	≈ 0	A few minutes	≈ 0 within field of application	> 10 min
START-UP TIME	> 2 sec	Top range : limited by warm-up	> 2 sec	> 1 min
AGEING	Low	Greater than (1)	No ageing within field of application	Same as (2)
MEASUREMENT PARTICULARITIES	Increased by drift in random flight	Numerous influencing elements	Testing facilitated by analogy in angular vel. and linear accel.	Very easy : no disassembly required for calibration
INDUSTRIAL ENGINEERING CAPABILITY	Proven, but remains difficult	Good	Good	Good

TABLE 6 - EXPECTED IMPROVEMENTS

CHARACTERISTIC	TECHNOLOGY		
	OPTICAL GYROSCOPE	WHEEL GYROSCOPE	INTEGRATED MECHANICAL SENSORS
MEASUREMENT RANGE	Very high capacity, presently limited by the interface and processing circuits	Compromises with performance	Very high capacity
BAND WIDTH, NOISE	Very high capacity (with reserve on locking zone rejection)	Very high performance for medium and low-performance ranges	Very high capacity
CONSUMPTION	Low	Depends on precision : reduced for low precision	Lowest of all
VOLUME	Minimal values are defined by the cavity length for the laser gyro (10 cm), or by the optical fibres diameter (several cm)	Miniaturization even more advanced (1 cm <sup>3</sup> )	On principle, volume is very low.

LES ARMES GUIDEES TIREES A DISTANCE DE SECURITE ET L'EVOLUTION DES MOYENS  
DE CALCUL EMBARQUE : L'APPORT DES TECHNIQUES NOUVELLES

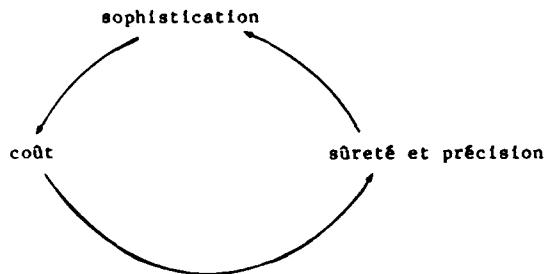
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Cet exposé présente les évolutions envisageables dans le domaine de la mise en oeuvre des armes tirées à distance de sécurité. Il détaille d'autre part les solutions techniques envisageables pour ce type d'armement dans les domaines du développement des logiciels et des matériels associés.

1. SITUATION DU PROBLEME

L'évolution considérable des moyens de défense aérienne interdit de plus en plus une pénétration efficace et une bonne approche des objectifs par un avion. Ceci a amené le développement d'un nouveau type d'arme dite "tirée à distance de sécurité". L'essor des moyens d'interdiction de zone oblige en effet à ne plus se contenter d'armes purement balistiques. Ces nouveaux concepts d'armes vont de la bombe à balistique améliorée (freinée puis accélérée par exemple) au missile de croisière, néanmoins le dénominateur commun de ces armements est une sophistication croissante qui en fait des armes chères dont on demandera alors d'être tirées à coup sur ce qui renforcera leur sophistication et ainsi de suite.



Il s'agit donc de briser ce cercle infernal en utilisant les moyens qui ont fait leurs preuves sur de gros systèmes d'armes. Ces armes nouvelles en effet mettent en oeuvre un nombre de plus en plus important d'équipements complexes :

- sur le vecteur tireur :  
(nacelle de désignation, centrale inertielle, visualisations, etc...)
- sur l'arme elle-même :  
(capteur sophistiqué, moyens de contre-mesure, etc...).

Ces équipements nécessitent des conduites de tir très élaborées avant et après tir :

- avant tir : il s'agit de désigner l'objectif, de transmettre à l'arme les paramètres nécessaires à la mission et ceci sans accroître de façon trop considérable la charge de travail du pilote, ce qui implique une mise en oeuvre particulièrement conviviale.
- après tir : il s'agit pour l'arme de gérer les équipements mis à sa disposition indépendamment du vecteur tireur selon le principe des armes "tire et oublie" là aussi, la complexité des phases de vol devient de plus en plus grande, et les logiques à mettre en oeuvre sophistiquées.

Le développement inéluctable de telles armes amène donc à tenter de résoudre les problèmes suivants :

- les logiciels de mise en oeuvre sont de plus en plus gros, leur maîtrise devient difficile,
- en corollaire les puissances de calcul nécessaires sont importantes,
- les volumes d'informations nécessaires à l'accomplissement de la mission deviennent considérables.

Nous allons voir dans la suite de l'exposé que les techniques nouvelles permettent de faire face à ces problèmes.

2. LE DEVELOPPEMENT DU LOGICIEL

2.1. INTRODUCTION

Les logiciels de mise en oeuvre des armes tirées à distance de sécurité sont des logiciels auxquels il sera demandé une très grande qualité car il n'est pas acceptable qu'une mission soit ratée par suite d'une défaillance de logiciel. D'autre part, l'augmentation de la taille des logiciels est inéluctable, nous en donnerons comme exemple les tailles des logiciels implantés

dans les calculateurs de missions embarqués sur avion :

Années : 78 - 80	de 30 à 80 Kmots de 16 bits
Années : 80 - 82	de 60 à 120 Kmots de 16 bits
Années : 82 - 85	de 100 à 300 Kmots de 16 bits

L'augmentation de la taille des logiciels embarqués traduit entre autres deux phénomènes :

- la complexité croissante de mise en oeuvre d'armes nombreuses et sophistiquées,
- l'amélioration du dialogue homme-machine afin d'alléger la charge de travail du pilote.

Cette taille croissante des logiciels de mise en oeuvre d'armement se retrouve d'autre part dans le graphique suivant qui donne pour trois types d'armements envisagés les volumes de logiciels nécessaires à la mise en oeuvre sur le vecteur tireur et sur l'arme elle-même.

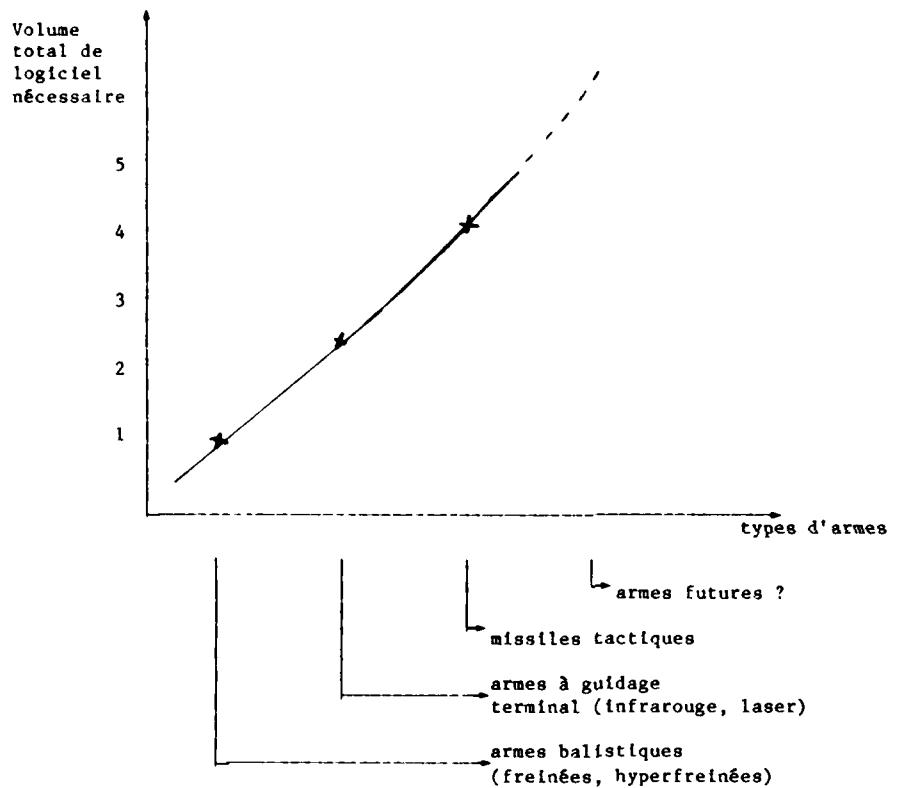


Figure 1

Les différents types d'armes portés en abscisse étant des armes à portée et à complexité de mise en oeuvre croissante.

Cette inflation du logiciel implique pour qu'elle soit maîtrisée l'utilisation de techniques de pointe :

- emploi de langage de haut-niveau autour d'un atelier de génie logiciel,
- appel aux techniques d'intelligence artificielle.

## 2.2. L'ATELIER DE GENIE LOGICIEL

Le logiciel à développer devra satisfaire à un grand nombre de facteurs de qualité :

- correction,
- robustesse,
- maintenabilité,
- adaptabilité,
- efficacité,
- etc...

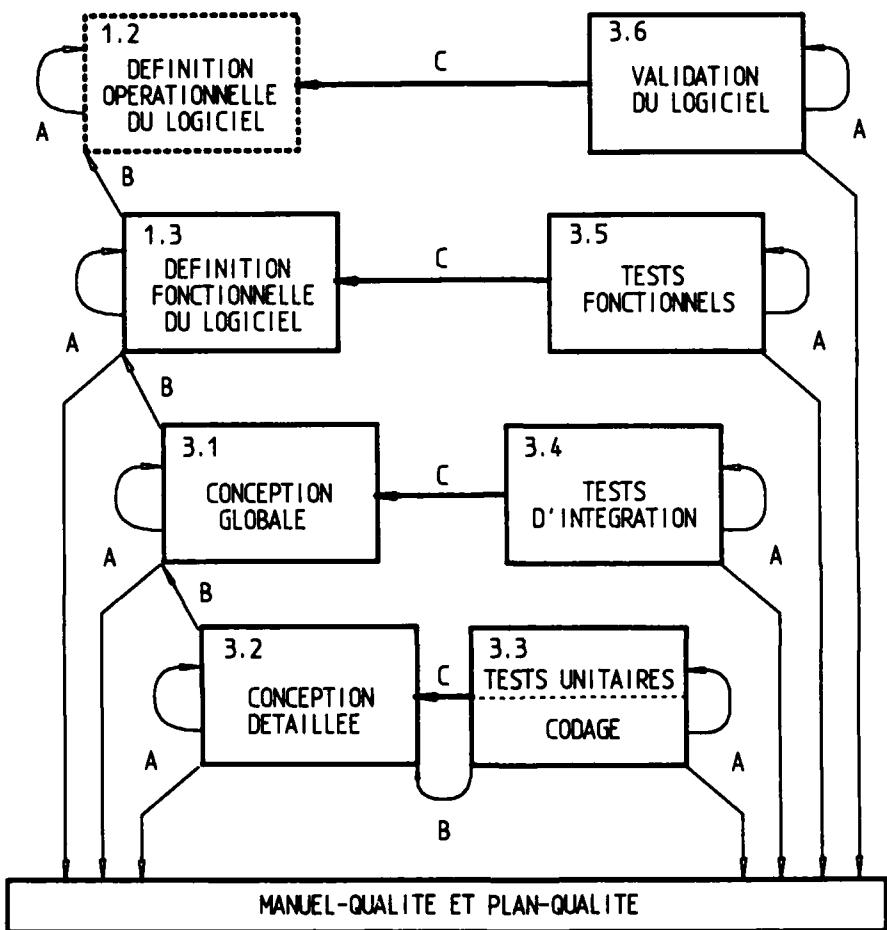
Cette qualité s'obtient par le contrôle d'un certain nombre de critères concourant aux facteurs de qualité :

- cohérence,
- complétude,
- tracabilité,
- modularité,
- simplicité,

- tolérance aux fautes,
- lisibilité etc...

Ce contrôle de la qualité s'obtient grâce aux manuel-qualité et plan-qualité élaborés en liaison stricte avec une méthodologie.

L'ESD a développé une méthodologie MINERVE dont l'assurance qualité repose sur ces principes :



En effet, à chaque étape sont faits un certain nombre de contrôles de trois types :

- Contrôles de type A :

ce sont des contrôles internes au produit d'une étape. Effectués par relecture et analyse des documents ou du code (selon l'étape concernée), ils consistent à vérifier que le produit de l'étape respecte les règles spécifiques (précisées dans le plan-qualité logiciel) et les standards généraux (définis dans le manuel-qualité logiciel) qui lui sont applicables dans le cadre du projet concerné (présentation de la documentation, règles de codage,...). On s'assure également de la cohérence interne, de la complétude, de la lisibilité et de la précision de chaque produit.

- Contrôles de type B :

ce sont des contrôles de cohérence entre les produits d'une étape et ceux des étapes antérieures. Ce type de contrôle, comme le précédent, est réalisé sous forme de relectures de documents et de revues de projet.

- Contrôles de type C :

ce sont les tests des programmes. Ils se déroulent en quatre étapes successives et s'effectuent à chaque étape selon des points de vue et par référence à des documents différents. Ainsi les tests réalisés au cours des étapes de codage et tests unitaires (3.3), tests d'intégration (3.4), tests fonctionnels (3.5), validation du logiciel (3.6) permettent de vérifier la conformité des programmes à leurs descriptions successives établies au cours des étapes symétriques : conception détaillée

lée (3.2), conception globale (3.1), définition fonctionnelle (1.3), définition opérationnelle (1.2).

Les tests ne constituent qu'un complément des contrôles de types A et B et ne doivent pas être considérés comme le seul moyen de contrôler la qualité du logiciel. Le passage d'une étape à la suivante est conditionné par l'obtention de résultats satisfaisants pour ces trois types de contrôle, schématisés sur la figure ci-dessus.

L'obtention d'une bonne qualité sera facilitée par l'utilisation d'outils informatiques organisés sous forme d'un atelier intégré de génie logiciel.

L'ESD a développé un tel atelier appelé "AIGLON".

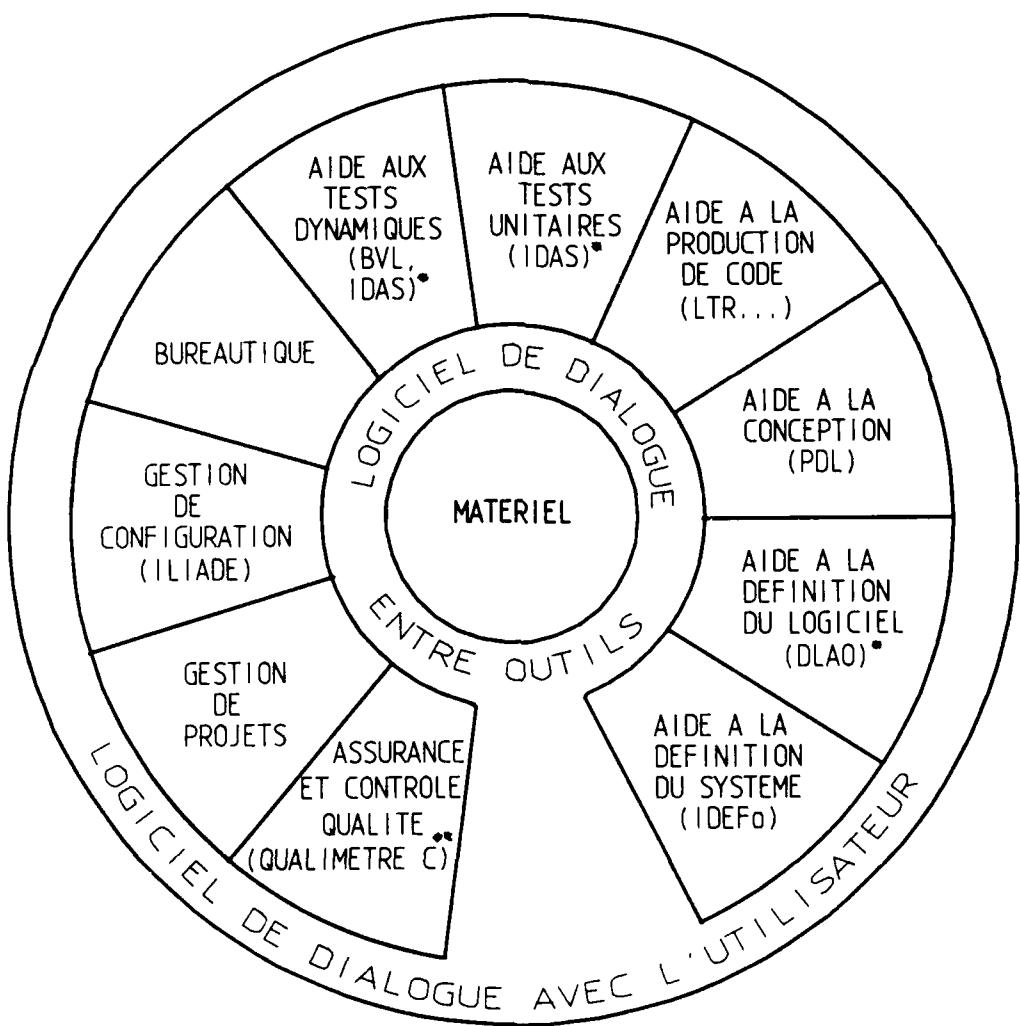


Figure 2 l'Atelier AIGLON

(\*) DLAO et IDAS sont des marques déposées à ESD  
 (\*\*) qualimètre C est une marque déposée IGL

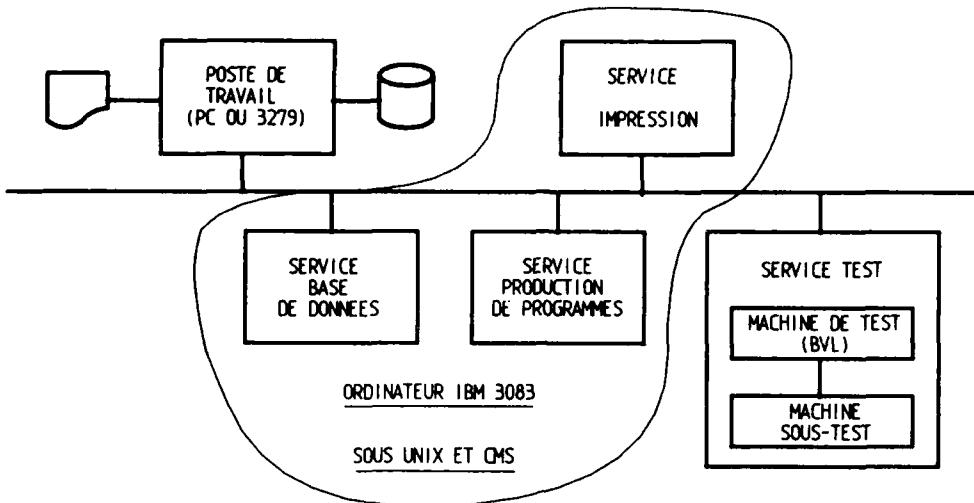


Figure 3 AIGLON : Architecture fonctionnelle

L'emploi d'un atelier intégré de génie logiciel et de langage de haut niveau (LTR3, ADA) facilite la maîtrise du coût et de la qualité des logiciels.

### 2.3. LES TECHNIQUES D'INTELLIGENCE ARTIFICIELLE

Le développement des armes tirées à distance de sécurité nécessitera, dans un avenir proche afin d'accroître les possibilités de telles armes, l'appel aux techniques d'intelligence artificielle.

En effet de telles techniques permettent de concevoir des systèmes où la conception et la réalisation se font non plus par méthode algorithmique mais par l'utilisation de règles. Ceci permet d'envisager une participation plus active des spécialistes non informaticiens.

Le recours à ces méthodes peut s'envisager dans deux voies :

- au niveau de la préparation et de la mise en oeuvre de l'arme afin de concevoir un système où l'arme "dialogue" avec le pilote de façon particulièrement conviviale, ceci afin d'alléger de façon significative la charge de travail du pilote.

En effet la part du logiciel consacré au dialogue homme-machine représente dans le cas de missiles tactiques jusqu'à plus de 50 % du logiciel total développé.

Grâce aux techniques d'I.A on pourra concevoir des systèmes où le pilote pourra interroger son SNA qui lui répondra les choix possibles en fonction des paramètres avion, carburant, emports, etc...

- à bord de l'arme elle-même après tir, de telles techniques ont des domaines d'applications multiples (navigation intelligente, reconnaissance de forme pour détection de menaces ECM ou recalage automatique, reconfiguration de systèmes en vol, ...).

L'utilisation de l'intelligence artificielle permettra sans aucun doute d'accroître de façon sensible les performances des armes tirées à distance de sécurité.

Enfin le recours à la commande vocale permettra certainement de simplifier certaines phases délicates de la conduite de tir en supprimant des insertions numériques, des appuis sur des touches, etc...

### 3. LES MOYENS DE CALCULS A METTRE EN PLACE

#### 3.1. INTRODUCTION

Les logiciels à implanter sur de telles armes nécessiteront des moyens de calculs puissants, permettant de faire des calculs classiques et symboliques (à cause de l'intelligence artificielle).

Nous illustrerons notre propos par deux graphiques :

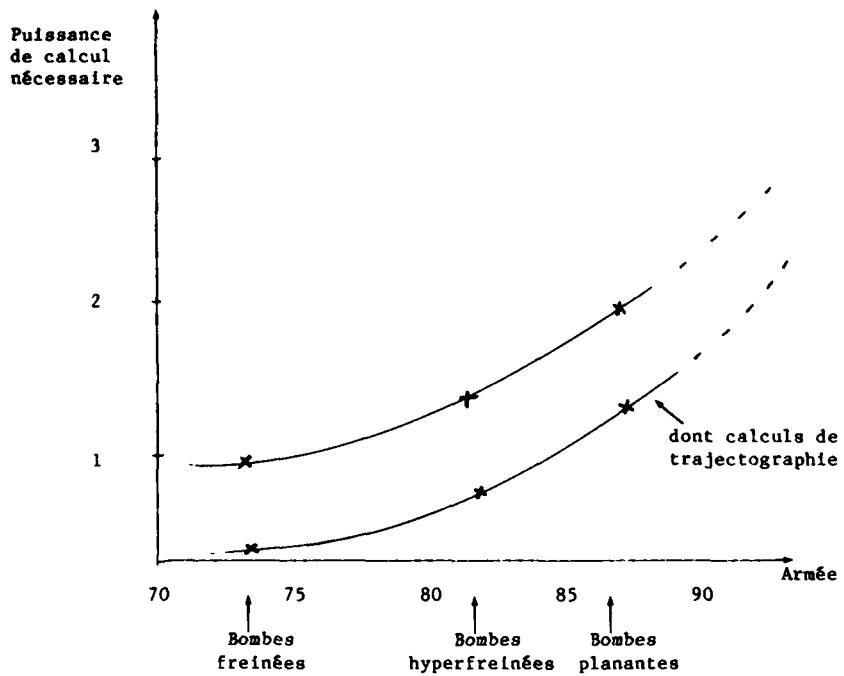


Figure 4

Le premier graphique (figure 4) donne pour un type d'arme : les armes balistiques, en fonction du temps et donc des évolutions techniques la puissance de calcul nécessaire à la mise en oeuvre et au tir de l'arme. Il donne d'autre part la puissance de calcul nécessitée par les calculs de trajectographie.

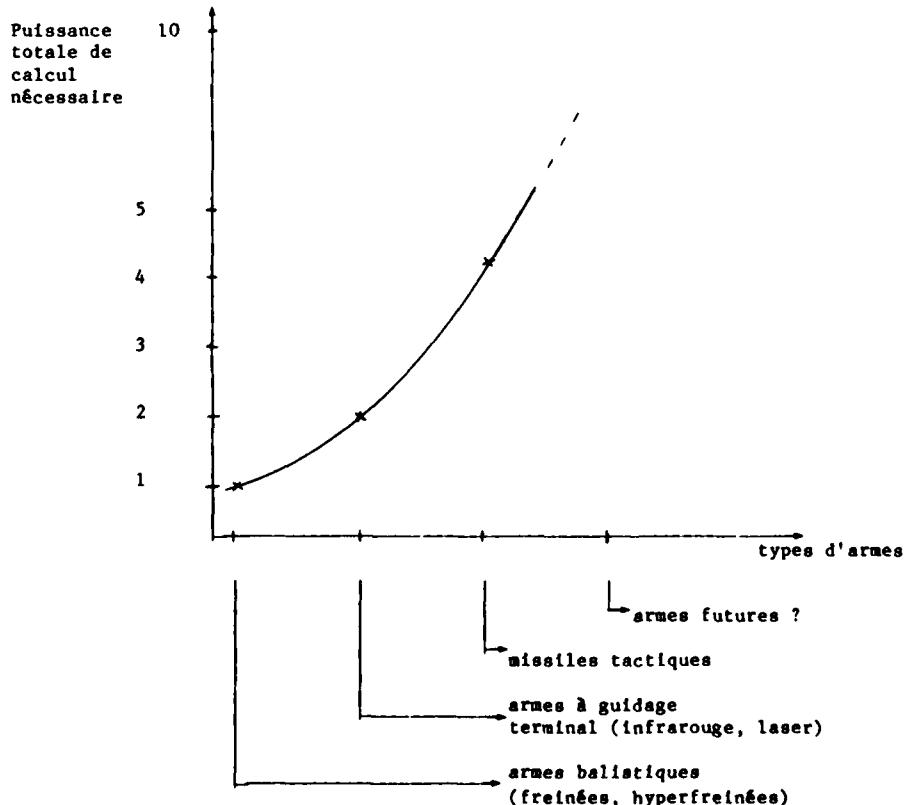


Figure 5

Le deuxième graphique (figure 5) donne avec le même système d'abscisses que sur la figure 1 les puissances de calcul totales à installer sur l'avion et sur l'arme pour la mise en oeuvre des armes :

- balistiques,
- à guidage terminal,
- missiles tactiques.

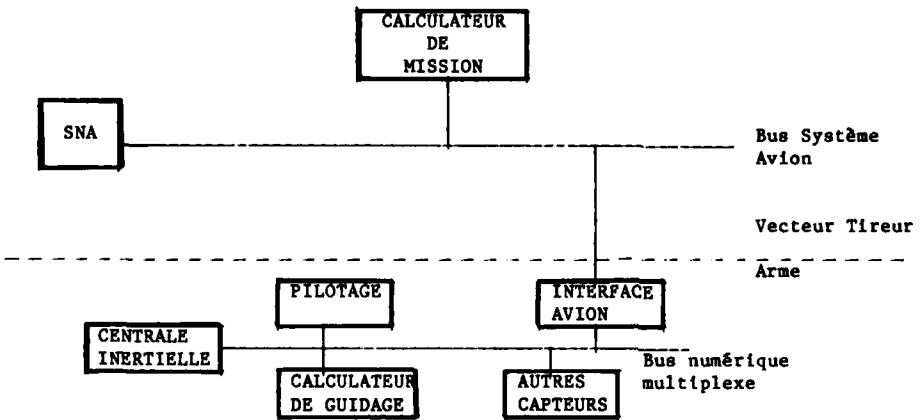
Les deux graphiques sont clairs que ce soit pour :

- améliorer des types d'armes existantes,
- concevoir de nouvelles armes,

le besoin en puissance de calcul s'accroîtra de façon considérable.

### 3.2. ARCHITECTURE POSSIBLE

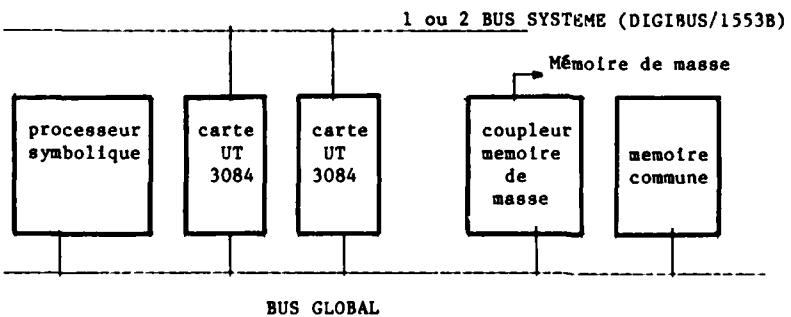
Les systèmes développés jusqu'à présent sont relativement classiques :



Les grosses évolutions porteront sur la structure des calculateurs, calculateur de mission ou calculateur de guidage afin d'améliorer :

- la puissance : utilisation de calculateurs multiprocesseurs
- le coût : standardisation des processeurs
- la sécurité et la fiabilité : redondance des processeurs et reconfiguration possible en vol

Une architecture possible développée par l'ESD pour le calculateur (de mission ou de guidage) est la suivante :



Chacune des cartes UT 3084 (1 Mops) est dotée de sa propre alimentation de son coupleur d'Entrée-Sortie de sa mémoire locale propre (256 à 512 Kbytes de 16 bits) et est entièrement banalisée, afin de faciliter les reconfigurations en cas de panne d'une des UT par exemple.

Le processeur symbolique permettra de développer des systèmes experts embarqués :

- de préparation et de reconfiguration de mission,
- d'aide au déroulement de la mission,
- de diagnostic de panne,
- etc...

#### 4. UTILISATION DE MEMOIRE DE MASSE

##### 4.1. INTRODUCTION

Les logiciels tels qu'ils ont été décrits précédemment impliquent la manipulation d'un grand volume de données d'où l'emploi de mémoires de masses embarquées.

##### 4.2. DONNEES A STOCKER

Un grand nombre d'applications nécessite le stockage de gros volumes d'informations, nous n'en donnerons ici qu'une liste non limitative :

- recalage par corrélation d'altitude ou cartographique : on stockera des terrains numérisés.
- reconnaissance de menace ECM : on stockera des bibliothèques de menaces.
- gestion de situations tactiques : enregistrement avant le vol ou en vol de renseignements tactiques : objectifs, type de défense, météo, ...
- enregistrement de paramètres en vol, on couvrira alors deux domaines d'applications le stockage de paramètres pour le développement et les essais ou bien dans le cas d'engin de reconnaissance type RPV le stockage des informations de reconnaissance à destination par exemple d'un moyen sol de restitution de mission.
- reconfiguration de mission en vol : on pourra stocker les programmes d'application qui seront appelés en overlay.
- etc...

Une rapide évaluation des volumes d'information permet de dimensionner les mémoires de masses nécessaires :

- cartes mémorisées pour recalage par corrélation (cartographique ou d'altitude)	100 à 500 Kmots
- bibliothèque de menaces	10 Kmots
- situations tactiques	10 à 50 Kmots
- enregistrement de paramètres en vol	10 à 100 Kmots
- programmes de missions pour reconfiguration en vol	100 à 500 Kmots

On obtient des volumes de mémoires à accéder en lecture et en écriture (au sol ou en vol) de l'ordre de 1 Mmot de 16 bits.

##### 4.3. LES SOLUTIONS TECHNIQUES

De tels volumes d'informations nécessitent des mémoires de masses fiables peu chères et faciles de mise en oeuvre :

Les mémoires de masses magnétiques paraissent mal adaptées à ce genre de problème, par contre les mémoires à semi-conducteur type EEPROM offrent toutes les garanties nécessaires y compris quant au temps d'accès qui pour certaines données est critique.

Le conditionnement de ces mémoires sous forme de cassettes enfichables permet d'autre part d'injecter rapidement dans le système un gros volume d'informations issues par exemple de la préparation de mission.

#### 5. CONCLUSION

Le développement des systèmes de défense aérienne induira un accroissement considérable du recours aux armes tirées à distance de sécurité. Ces armes seront de plus en plus sophistiquées, les moyens techniques pour les développer existent ou sont en plein essor comme l'appel aux ressources de l'intelligence artificielle et sont bien adaptés aux problèmes à résoudre.

Precision Delivery of Unguided Submunitions  
From a Tactical Standoff Missile

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SUMMARY

The extension of a low cost tactical weapon to missions involving area targets requires the accurate dispensing of unguided submunitions. For a terminally guided weapon, proportional navigation can be no longer be used, since it cannot compensate for the ballistics of the submunitions. A novel terminal guidance algorithm has been developed to control the weapon trajectory between the midcourse flight path and the point of dispense. The algorithm incorporates submunition drag characteristics to assure high accuracy. The algorithm also assures adequate arming time for the submunitions and adequate time for the pattern to develop. The second element of accurate submunition delivery is the dispensing or expulsion mechanism, which ejects the payload out of the canister. The ejection mechanism must provide the desired pattern size, shape and density. Air bag technology is employed to assure satisfactory ejection forces with adequate repeatability. The current study is theoretical in nature, relying of digital simulations to substantiate conclusions on patterns and accuracy.

1.0 INTRODUCTION

The investigation of highly accurate dispensing of unguided submunitions from a tactical standoff missile was conducted as part of an Air Force funded study (F08635-84-C-0255) to develop a rocket propelled version of the GBU-15 glide bomb. Two warheads were included in this modular concept, a unitary warhead (MK-84 bomb) for hard point targets and a SUU-54 dispenser loaded with submunitions for softer area targets. The other elements of the weapon system are the airframe, flight control and guidance modules. The system utilizes either a TV or IR guidance module. The weapon is controlled after release by a weapons system operator (WSO) through a data link. The WSO adjusts heading and altitude and acquires the target during midcourse flight. The unitary warhead version employs proportional navigation for terminal guidance. The submunition dispenser version utilizes trajectory shaping and a dispense algorithm to place submunitions in a circular pattern around the aimpoint viewed by the WSO. The weapon system with unitary warhead, now designated the AGM-130A, is currently under full scale engineering development by Rockwell International. At the same time, the dispenser version, AGM-130B, is undergoing further analysis with various submunition payloads.

The concept for delivery of unguided submunitions involves two basic elements. The first is a terminal guidance algorithm to control the weapon trajectory between the midcourse, or hold altitude, flight path to the point of dispense. This algorithm must compensate for the ballistics of the submunitions, which commonly employ a high drag device for providing high impact angles. The algorithm must also assure adequate arming time for the submunitions and adequate time for the pattern to develop. The second element of submunition delivery is a dispensing or expulsion mechanism, to eject the payloads out of the canister. The ejection mechanism must provide the desired pattern size, shape and density. This paper describes the analysis and simulation of both of these basic elements.

The terminal guidance algorithms developed incorporate an Extended Kalman Filter (EKF) to combine the measurements from the terminal seeker and low cost inertial sensors to estimate the required system states. A microprocessor-based digital controller is used to implement the optimal estimation (EKF) technique, the guidance algorithm and other autopilot functions. The guidance algorithm is of particular interest, since unlike other terminal guidance laws, the algorithm must simultaneously control both the final position and velocity of the delivery vehicle. Elements of this algorithm include: tabular data of submunition ballistics, trajectory control between midcourse altitude and optimum dispense point, and automatic control of the dispense sequence as the optimum dispense point is approached. Simulation of the weapon and submunitions indicated the concept to be robust in the presence of variations in midcourse speed and altitude, terminal dive angle, and weapon bandwidth. A preliminary error analysis identifies the major sources of error, and their effect on system accuracy. Accuracy, in this case, is defined in terms of the displacement of the centroid of the pattern from the desired aimpoint.

The dispensing mechanism developed was the result of extensive simulation studies and coordination with airbag manufacturers. The resulting ejection concept incorporates a dual airbag system in which individual submunitions are grouped into subpacks consisting of submunitions, an airbag and foam dunnage. These subpacks are arranged around one or more primary airbags which are used to eject the subpacks radially away from the main canister to begin the dispensing sequence. After a prescribed delay, the secondary airbag in each subpack is activated providing a pattern of the desired size and density. The pattern optimization problem involves the simultaneous fulfillment of a number of conflicting goals or requirements. These include operation over a range of dispense conditions (speed, altitude, dive angle), providing a minimum time of flight for submunition arming and developing the desired pattern size, shape and density to

maximize the probability of kill for the range of targets specified. Through simulation, the sensitivity of pattern shape to a number of parameters was determined. These parameters include: ejection velocities, dispense speed and dive angle, and dispense altitude.

In summary, the extension of a low cost tactical weapon to missions involving area targets is developed. The study is theoretical in nature, relying on simulation to substantiate conclusions on patterns and accuracy. However, the simulation models are the result of numerous flight tests with the GBU-15 weapon system. The dispensing algorithms and ejection mechanisms are easily adapted to other submunitions and target types.

## 2.0 BACKGROUND

The GBU-15 is an operational glide bomb currently in the inventory of the United States Air Force. The weapon consists of a MK-84 (2000 lb) bomb with conventional explosives, which is guided to the target by an electro-optical (TV) or infrared seeker. The seeker signals combined with attitude gyro and accelerometer outputs drive a pneumatic actuator for control of aerodynamic surfaces or fins. Forward mounted strakes and aft mounted cruciform wings provide lift. A data link is employed between the weapon and launch aircraft. This allows the weapon systems operator (WSO) to control the weapon trajectory and to acquire and lock-on to the target by monitoring the seeker imagery. The tracking of the target can be accomplished either automatically or manually by the WSO. The weapon is designed for direct hits against hard targets such as bridges, buildings and SAM sites. The weapon is carried by two seat aircraft such as the F-4E and the F-111.

A pre-planned product improvement effort was initiated in 1983 (Reference 1) to increase the standoff range and thus improve the delivery aircraft survivability. This extended range required the addition of a solid propellant rocket motor. A radar altimeter was added to allow low altitude flight in a terrain avoidance or hold altitude mode. The aft lifting surfaces or wings were re-configured to permit the use of dispenser warheads. Reference 1 also includes the results of a study to dispense BLU-97/B or CEB submunitions from a SUU-54 dispenser for the attack of soft, area targets.

In the fall of 1984, Rockwell International began full scale engineering development of this improved GBU-15 with MK-84 warhead which was designated AGM-130A. The development effort, funded under U.S. Air Force contract F08635-84-C-0255, also provided for further investigations of submunition dispensing. These investigations (Reference 2) utilized the BLU-106/B (BKEP) and HB876 (Mine) submunitions in the inventory SUU-54 dispenser. This version was designated AGM-130B. The objective of the AGM-130 is to provide airfield attack capability while retaining the AGM-130 system standoff range for delivery aircraft survivability.

## 3.0 BASIC CONCEPTS

The purpose of this section is to describe the common characteristics of the unitary warhead (AGM-130A) and dispenser (AGM-130B) versions of the weapon. Then the differences will be discussed as they relate to terminal guidance and dispensing methods. The AGM-130 weapon configurations are shown in Figure 1. Major subsystems such as warhead, seeker, sensors, rocket motor, etc. are identified. All of these subsystems are common to both versions with the exception of the warhead and forward and aft castings which mate the seeker and control sections with the warhead. The digital processor, which is also common to both versions, uses an input signal to identify the warhead configurations. This signal selects between two sets of software, one for each of the two missions.

The basic flight profiles are presented in Figure 2. The missions begin with a climb or glide to a commanded midcourse altitude after separation from the launch aircraft. After stabilizing at the midcourse or cruise altitude, the rocket motor will fire when the estimated dynamic pressure drops below a preset threshold. During the midcourse phase, the weapon systems operator acquires the target by using the video images transmitted from the missile. The seeker or operator then tracks the target to impact for the AGM-130A or to the dispense point for the AGM-130B. The terminal trajectories for the two systems differ markedly due to the differences in terminal guidance and system drag. These differences are discussed in the following sections.

### 3.1 Guidance Concepts

There are several fundamental differences between terminal homing to impact for unitary warheads and the guiding of a submunition canister to a dispense point. The geometry of the dispensing situation is shown in Figure 3. For a standoff weapon, there is normally a fairly long midcourse phase, followed by a terminal maneuver. The terminal maneuver ends at a burst or dispense point at which the canister opens and the submunitions are released. The ejection mechanism imparts forces normal to the flight path so that a wide pattern of ground impacts is created. Typically each submunition has a high drag device such as a parachute or ballute to arm the warhead and assure near vertical impacts.

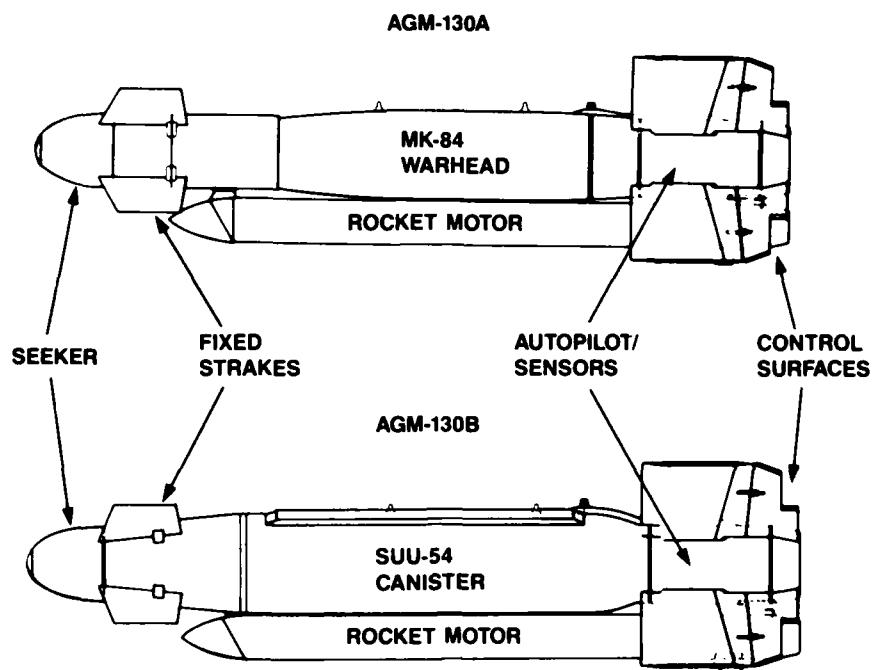


Figure 1. AGM-130 Configurations

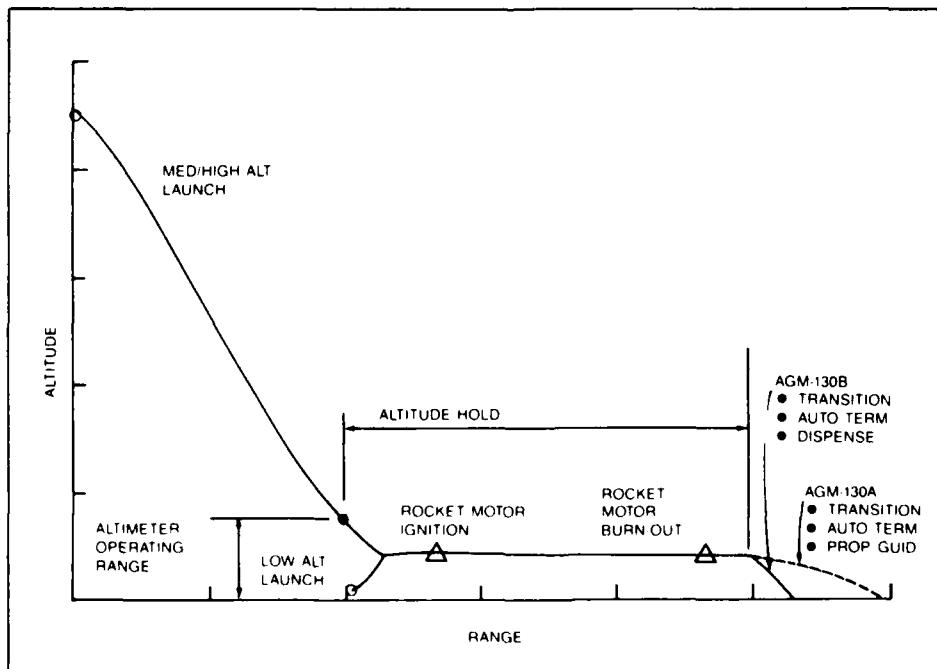


Figure 2. AGM-130 Flight Profiles

The difference in terminal guidance to impact and guidance to dispense point are summarized in Table I. As one might expect, the accuracy required for a unitary warhead is much greater than for a submunition dispenser. The radius of damage for a dispenser may be an order of magnitude larger than the radius of damage for a unitary warhead. In general, the required CEP will be proportional to the effective radius of damage. This is fortunate, since the accuracy with which submunitions can be delivered is considerable lower than for payloads guided to impact. The relatively long period of unguided flight (typically 3-6 seconds) between dispense point and impact make the system sensitive to errors. Winds, dispense timing errors, and delivery errors are generally the most serious errors. This sensitivity is aggravated by the high drag of the submunition, which may be an order of magnitude greater than that of the delivery vehicle itself.

Another significant difference in guidance strategy is that the dispensing weapon requires control of both the position of the burst point (i.e., altitude and downrange distance) and the flight path angle. Control of the flight path angle is not required for high accuracy of single payloads. In addition, these dispense conditions:  $h_{HoB}$ ,  $x_{HoB}$ , and  $\gamma_{HoB}$  (Figure 3), are functions of the weapon speed at dispense ( $V_{HoB}$ ). This can be recognized from the fact that the flight of the submunitions is essentially ballistic. Therefore the trajectory can be described by a set of differential equations of a point mass acted on by aerodynamic forces (drag), gravity, and perhaps thrust. The force equations can be integrated twice to yield the trajectory after the appropriate initial and terminal conditions have been applied. The result is that the downrange distance between the burst point and the impact point ( $x_{HoB}$ ) is only a function of the altitude ( $h_{HoB}$ ), speed ( $V_{HoB}$ ), and flight path angle ( $\gamma_{HoB}$ ) at dispense. Therefore, this function ( $f$ ) equals:

$$X_{HoB} = f(\gamma_{HoB}, h_{HoB}, V_{HoB})$$

This functional form suggests that ballistics tables can be generated as a function of the three independent variables. Since the dimension of the table is only three, and if the range of each variable is not large, the storage requirements for an on-board processor would be reasonable. This is one element of the dispensing algorithm to be presented in a later section. A related element is that knowledge of the four system states must be available within the weapon. For the AGM-130B weapon, a radar altimeter, autopilot grade inertial sensors (i.e., gyros and accelerometers) and the terminal seeker provide the necessary inputs to compute the system states. An Extended Kalman filter (EKF) is employed to combine the measurements in an optimal manner. Only a single plane representation of weapon motion is required. This is motion in the vertical plane. The yaw or crossrange motion is controlled by a proportional navigation guidance law.

Another element of the guidance concept involves system constraints. Submunition arming requirements and minimum flight times needed to create the desired pattern size restrict the dispense conditions. Another constraint results from the limited tracking rate of the weapon seeker. In general, the longer the seeker is able to track the target, the more accurate will be the system state estimates described above.

### 3.2 Dispensing Methods

As previously mentioned, the dispenser version of the weapon system utilizes the SUU-54 A/B canister, which is capable of dispensing a variety of submunitions, depending on the desired application. Many submunitions utilize a high-drag device such as a parachute or ballute (short for "balloon parachute") to slow the submunition so that gravity can turn the terminal flight path to a nearly vertical orientation. Some kinetic energy penetrators, such as the BLU-106/B (BKEP) submunition, use a rocket motor to boost the submunition's speed following the reorientation of the flight path to a high-speed, nearly vertical descent until impact in the target area. Others, such as the BLU-97/B (Combined Effects Bomblet, CEB) and HB-876 (Mine) retain the drag device for the remainder of their flight, which results in a low terminal velocity. The particular submunition selected for the dispenser version depends on the target class against which it will be used. Soft area targets, such as light vehicles, radar/communication vans, personnel, etc. are a common application for CEB-type submunitions, while hard area targets, such as airport runways or taxiways are a common application for BKEP-type submunitions.

Rockwell International has performed similar studies to develop dispenser systems which utilize both types of submunitions. The dispenser system developed for use for soft area type targets using CEP-type submunitions included 384 separate submunitions which were spread out over a circular pattern to cover a very large surface area. The basic pattern requirements (other than pattern size, which is determined by target threat/effectiveness analyses) are uniformity of the pattern so that there are no significant voids in the impact, or target area, and that the flight time of the submunitions is sufficient to allow for their warheads to arm. The dispenser system developed for use for airfield attack missions uses BKEP and MINE submunitions.

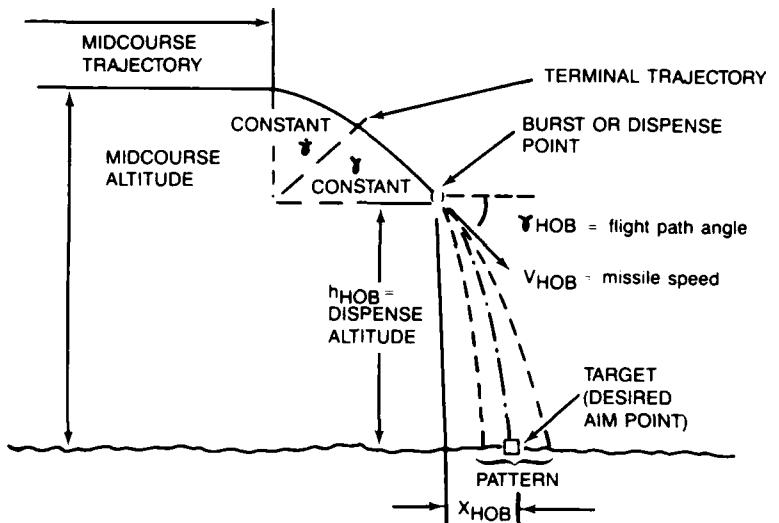


Figure 3. Dispensing Geometry

Table I. Differences In Guidance Concepts  
For Unitary And Disperser Warheads

Factors or Parameters	Terminal Guidance To Impact	Guidance to Dispense Point
(1) Required accuracy	high (small radius of damage)	low (large radius of damage)
(2) Target Type	hard, point target	soft and/or area target
(3) Length of unguided flight	small (1/2 to 1 sec)	large (3-6 sec. to go)
(4) Sensitivity to Errors	small (due to (3) and low drag (5))	large
(5) Drag during terminal guidance	low (speed approx. constant)	high (sensitive to winds)
(6) Control of flight path angle	no affect on accuracy	required for high accuracy
(7) Knowledge of system states	unnecessary	required

Because of space limitations in the SUU-54 canister and the length of BKEPs, only 15 BKEPs can be packaged in this version of the dispenser weapon, along with 60 MINES. The desired pattern size (again determined by target threat/effectiveness analyses) is roughly elliptical with dimensions of about 300 feet long by 200 feet wide. The basic pattern requirements are, again, uniformity of the pattern and sufficient flight time (this time for the BKEP rocket motor to burn out before ground impact). However, in this application, pattern uniformity requires both an even spacing of the submunition impacts, plus the added requirement of no clear path through the pattern greater than a specified length (sufficient to taxi an aircraft through). The pattern uniformity requirement for BKEPs is more significant than for the CEBs because of the limited numbers of BKEPs dispensed by the system.

The current dispenser weapon research at Rockwell International involves airfield attack missions utilizing the BKEP submunitions, along with simultaneous dispensing of HB-876 Mines to slow any repair operations following the attack. Further discussions in this paper will be limited to this version of this dispenser system.

The basic dispense/packaging concept involves packaging the submunitions into rigid foam dunnage around a smaller airbag/inflator system. These "subpacks" are then packaged into the canister around the primary airbag(s)/inflator(s) which are secured together with metal straps designed to shear at a pre-determined load. The dispense event begins at the time that the guidance algorithm determines that the proper relationship exists between weapon speed, altitude, flight path angle and distance from the target area. At that time, a pyrotechnic charge initiates a gas generator in the primary inflator to inflate the primary airbag, which expels the subpacks containing the submunitions. After a pre-determined time delay (which is in general a function of the subpack's pre-dispense position in the canister), the secondary inflators inflate the secondary airbags, which separates the subpacks, exposing the individual submunitions to the freestream airflow. Figure 4 illustrates typical packaging and dispense of the BKEP/MINES submunitions. The time-sequencing of the BKEP submunitions and the HB-876 Mines differs from this point on. The BKEPs have pop-out fins which are deployed for stability, and after a pre-determined time delay, a parachute is deployed. After another pre-determined time delay, the parachute is jettisoned and the rocket motor fires which greatly increases the submunition's speed until burnout just before impact. Following a short time delay after impact, the submunition explosive is activated, which heaves up the runway surface, causing a large crater. Figure 5 is an illustration of the BKEP submunition and its operational timeline. The Mines use a small drogue parachute instead of fins for stability, which is deployed at about 76 milliseconds after exposure to the freestream. Again following a pre-determined time delay, a retarder parachute is deployed, which greatly reduces the Mines velocity and turns the flight path to a near vertical orientation. As mentioned, the retarder parachute remains attached until impact, so that the Mines submunitions' terminal velocity is small. After a time delay, the mine releases the parachute, and an erection mechanism causes the mine to erect itself to a vertical, or standing position. Figure 6 is an illustration of the MINES submunition and its operational timeline. After a variable time delay (different for each mine to hamper mine removal efforts) the Mines arm themselves and remain so until activated. To avoid interference with the BKEPs, the Mines should impact the ground after the BKEPs have exploded, but should cover the same ground pattern to maximize their effectiveness in preventing runway repair.

The major technical problem with the BKEP/Mines system is attempting to accurately overlay the ground patterns of two groups of submunitions which have different ballistic characteristics. One possible solution is to perform a sequential, rather than simultaneous dispense. That is, the BKEPs can be dispensed first, while the dispenser continues flying for a short time at which time the Mines can be dispensed, or vice-versa. For the applications considered by the Rockwell International studies, the required time delay between the two sequential dispensing events is small so that the concept is feasible. Another possible solution is to dispense the submunitions simultaneously, but to use the basic time delays that each have in their flight profile timelines (for parachute deployment, etc.) to control their trajectories to meet the constraints mentioned above. In the studies conducted to date, both approaches have been considered; however, only the simultaneous dispense method will be discussed further, as it yields acceptable results and has less technical risk than the sequential dispense method.

#### 4.0 TERMINAL GUIDANCE ALGORITHMS

The algorithm which was developed for the guidance of a dispenser payload has a number of desirable features. First, it provides simultaneous control of both the position and flight path angle at the point of dispense. Secondly, the terminal guidance does not require maneuvers of high acceleration or high bandwidth from the vehicle. System accuracy is therefore almost independent of weapon characteristics. The algorithm development incorporates the system constraints, so that they will automatically be met. Finally, since the time-to-go to the burst point is continuously output by the algorithms, time delays due to fuzing, canister opening, etc. can easily be compensated for. The algorithm is easily adapted to various submunitions by modification of the ballistics tables and constraints equations.

The following sections describe the development of the dispensing algorithm, the incorporation of system constraints, and the effect on accuracy of various error sources.

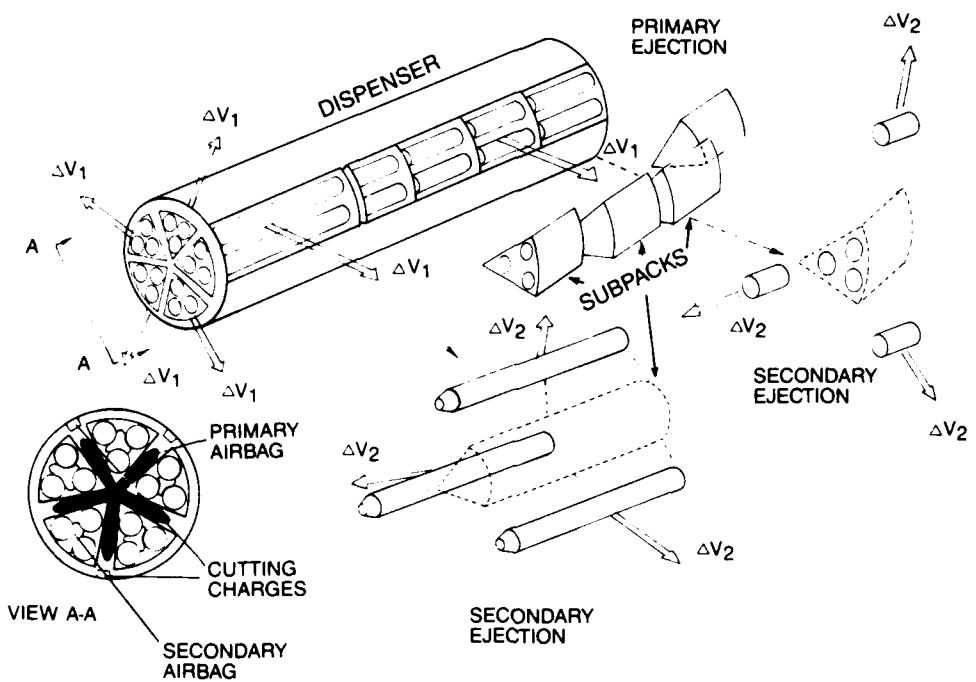


Figure 4. Illustration of Packaging/Ejection Concept

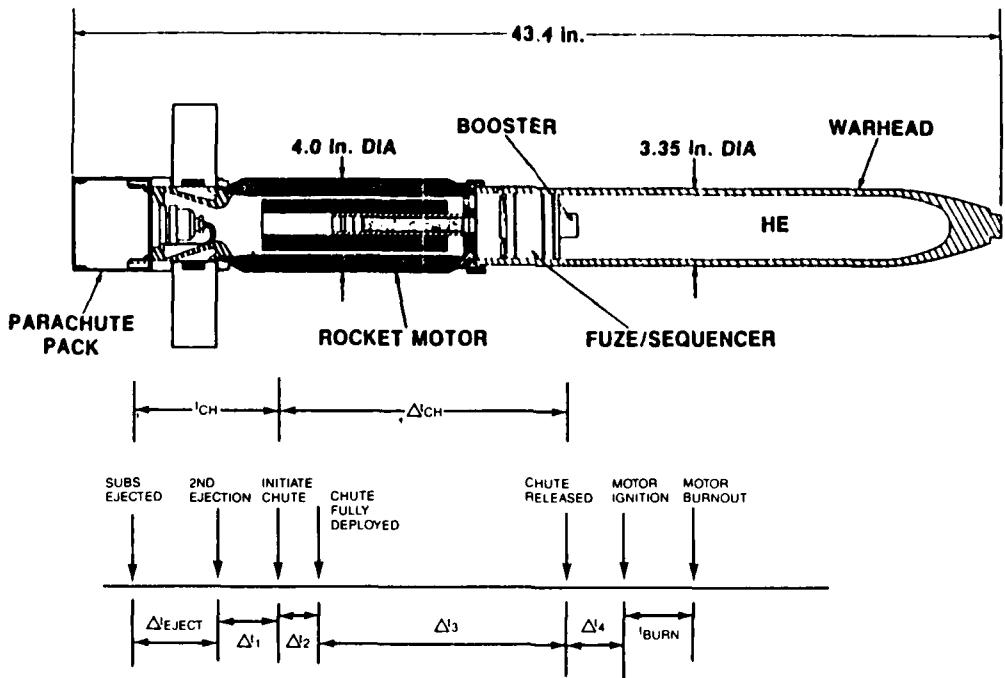


Figure 5. BLU-106/B (BKEP) Physical Characteristics and Operational Timeline.

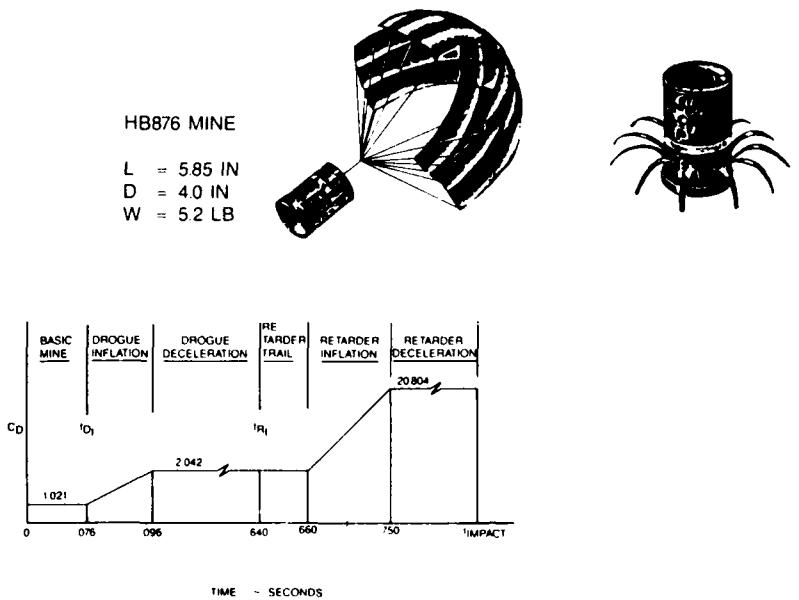


Figure 6. HB-876 (MINE) Physical Characteristics and Operational Timeline

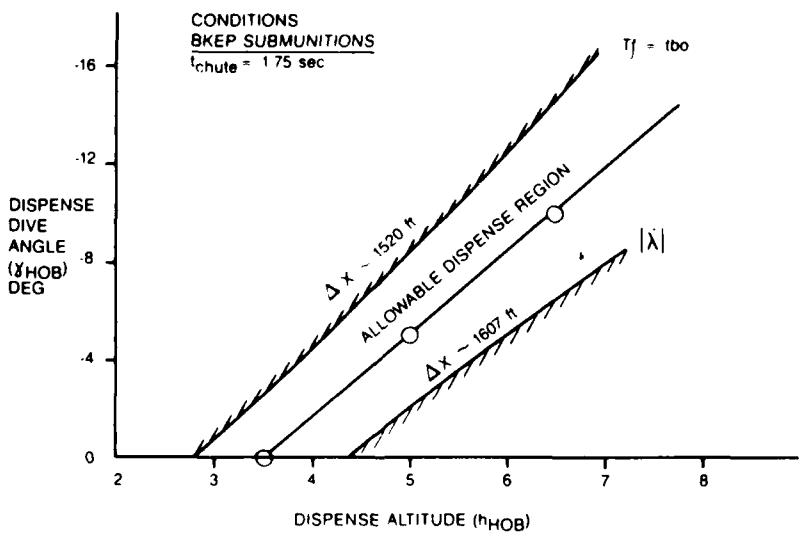


Figure 7. Constraints on Dispense Conditions

#### 4.1 Algorithm Features and System Constraints

Control of the terminal flight path through terminal guidance to the dispense point is provided by this algorithm. Information is received on the relative position and velocity of the missile with respect to the target from the Kalman filter described previously. To offset the effects of gravity on the submunitions, the optimal missile flight path is estimated such that the velocity vector is above the line-of-sight to the target at the dispensing point. The trajectory between the midcourse "hold altitude" phase and payload impact is shown in Figure 1. The transition between midcourse and the dispense point includes a constant acceleration ( $\ddot{\gamma} = \text{constant}$ ) phase followed by a constant flight path phase ( $\gamma = \text{constant}$ ). The dispense algorithm includes the following four steps:

- (1) selection of the optimum dispense point ( $V_{HOB}$ ,  $h_{HOB}$ ,  $\gamma_{HOB}$ ) for given acquisition or midcourse conditions ( $V_{ACQ}$ ,  $h_{ACQ}$ ).
- (2) selection of the optimum two phase trajectory between the midcourse altitude and the optimum dispense point.
- (3) computation of the time-to-go ( $T_{GO}$ ) to a zero miss distance condition using ballistic tables for the submunitions.
- (4) initiation of the dispense sequence at a critical time-to-go which compensates for the fusing delay, canister opening, and air bag operation.

The selection of the optimum dispense point for a given acquisition altitude and speed is based on the intersection of two straight lines in the  $\gamma_{HOB}$ / $h_{HOB}$  plane, where:

$\gamma_{HOB}$  = weapon dive angle at dispense

$h_{HOB}$  = weapon altitude at dispense.

It is in this plane that the system constraints can be displayed graphically. For the example shown in figure 7, the constraints are the burn-out of the submunition rocket motor before impact ( $T_f = t_{bo}$ ) and the seeker tracking rate limit. Within these boundaries is region of allowable dispense conditions. The first of the two lines mentioned above is the line midway between the boundaries. The equation for this line ( $L_1$ ) equals:

$$\gamma_{HOB} = m_y (h_{HOB} - h_b)$$

where  $m_y = 0.0697$

$$h_b = 335 \text{ ft. (102. m)}$$

The other straight line ( $L_2$ ) represents a terminal trajectory of constant dive angle ( $\gamma = \gamma_{HOB}$ ) with a specified time of guided flight. The equation for this line equals:

$$\gamma_{HOB} = \sin^{-1} \left[ \frac{h_{ACQ} - h_{HOB}}{T_g V_{ACQ}} \right] = \left[ \frac{h_{ACQ} - h_{HOB}}{T_g V_{ACQ}} \right] 57.3 \text{ deg.}$$

For the case shown in Figure 8, the selected dispense condition is designated "optimum dispense point". To limit the size of ballistics tables which must be stored in the memory of the digital controller, a set of 8 selected dispense conditions ( $h_{HOB}$ ,  $\gamma_{HOB}$ ) has been chosen. Therefore, the selected dispense conditions must be quantized. The quantized value that most nearly approximates the computed intersection is selected. The dispense conditions are also quantized with respect to missile speed, with the 4 values chosen.

The second step, that of selecting the optimum trajectory between the acquisition trajectory and the dispense point, employs the ballistics tables for the submunitions. The table yields a downrange distance,  $X_{HOB}$ , of the dispense point with respect to the target, of the form:

$$X_{HOB} = f (\gamma_{HOB}, h_{HOB}, V_{HOB})$$

Where

$\gamma_{HOB}$  = dive angle at dispense  
 $h_{HOB}$  = altitude at dispense  
 $V_{HOB}$  = speed at dispense

Since the dive angle and altitude are related for the nominal dispense points, then the table can be reduced from three to two dimensions. Let:

$$i = \text{dispense point altitude/dive angle index } (i = 1, 2, \dots, 8) \\ j = \text{dispense point velocity index } (j = 1, 2, 3, 4)$$

Then a set of 5 coefficients, i.e.,  $A_0(i,j) \dots A_4(i,j)$ , can be defined to provide accurate interpolation between the 32 tabular points  $(i,j)$ . The equation for the downrange distance,  $X_{HOB}$ , becomes:

$$X_{HOB}(\gamma, h, V) = A_0(i,j) + A_1(i,j) \Delta V + A_2(i,j) \Delta \gamma + A_3(i,j) \Delta \gamma^2 + A_4(i,j) \Delta h$$

$$\begin{aligned} \text{where } \Delta V &= V - V_j \\ \Delta \gamma &= \gamma - \gamma_j \\ \Delta h &= h - h_j \end{aligned}$$

This yields a table of 160 values, which for reasonable errors from the tabular points, has an rms error on the order of 1.0 to 2.0 feet. The guidance law for the terminal phase is to hold the desired course line, which is similar to the hold altitude law except that the reference line is tilted from  $\gamma = 0$  to  $\gamma = \gamma_{HOB}$ .

The third step is to compute the time-to-go ( $T_{GO}$ ) to the optimum dispense point during terminal guidance. This is done by computing the intersection of the extrapolated trajectory with the equation of acceptable dispense conditions based on the ballistics tables. This equation has the form:

$$X_{HOB} = X'_{HOB}(V_j, \gamma_{HOB}) + a_4(\Delta h)$$

$$\text{Where } a_4 = \frac{\partial x}{\partial h}$$

$$\Delta h = h_1 - h_2$$

Then, the three equations to be solved simultaneously for  $T_{GO}$  are:

- (1)  $X^* = X_E + V_{XE} T_{GO}$
- (2)  $Z^* = Z_E + V_{ZE} T_{GO}$
- (3)  $X^* = X'_{HOB}(V, \gamma, h_{HOB}) + a_4(-Z^* - h_{HOB})$

Where  $X^*$  = downrange distance to dispense point.  
 $Z^*$  = vertical distance to dispense point.

The result equals:

$$T_{GO} = \left\{ \frac{(X'_{HOB} - X_E) + a_4(-Z_E - h_{HOB})}{(V_{XE} + a_4 V_{ZE})} \right\}$$

Step four is the initiation of the dispense sequence as  $T_{GO}$  crosses zero. If considerable delays are found in the fusing process, i.e., release of SUU-54 panels, activation of air bags, etc., then this can easily be compensated for by using a  $T_{GO}$  other than zero to begin the dispense process.

A typical simulated trajectory is shown in Figure 9. Also shown is a reference or commanded trajectory. The midcourse phase was a hold altitude mode with an altitude command of 2000 feet (609.6 meters). The weapon then executes a 3g turn for 4 seconds. Since the speed is approximately Mach 0.6, the flight path turn rate ( $\gamma$ ) is a constant 2.8 degrees/sec. The system then follows a flight path with a constant dive angle of -20.6 degrees. The dispense point is 650 feet (200 meters) above the terrain, 1178 feet (359 meters) downrange of the target. The impact dive angle is roughly -60 degrees due to the high drag of the submunitions. The locus of dispense conditions for zero miss distance is shown for the nominal speed. Dispensing occurs when the estimated weapon position crosses this line. The guidance algorithm replaces guidance to a specific point, with dispensing when the flight path crosses the zero miss locus.

#### 4.2 Accuracy Considerations

The accuracy of the delivery of submunitions can be measured in terms of the distance between the centroid of the submunition ground impacts and the desired aimpoint. From the simulation, this impact centroid can be computed by simulating all the submunitions or by simulating a submunition with no ejection forces. Using the later method, the accuracy of algorithm is such that the miss distances for simulated trajectories of typical submunitions without system errors are less than two feet as shown in Figure 10. The miss distances are attributed to finite simulation step sizes and ballistics table errors. This plot demonstrates the robustness of the concept with regard to the dispense speed and altitude. This also indicates that the concept does not rely on the weapon system dynamics, but is sensitive to dispense timing errors.

The other significant error sources are listed in Table II for the case of dispensing BKEP warheads and Mines. Timing errors, whether in fusing, chute deployment or canister opening result in large errors in downrange distance. A 5.0 percent error in submunition weight has about the same effect as a 5.0 percent error in drag. Since the payloads are unguided after ejection from the SUU-54 canister, they are subject to winds.

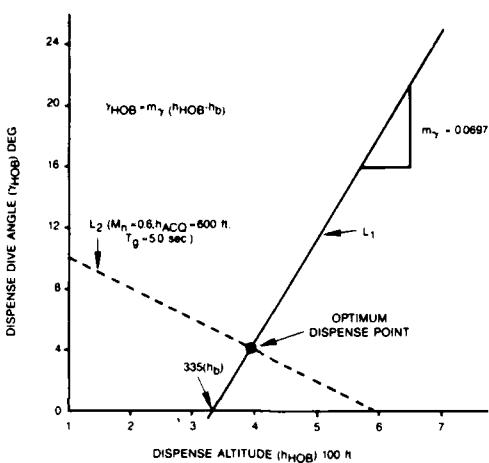


Figure 8. Selection of Optimum Dispense Point

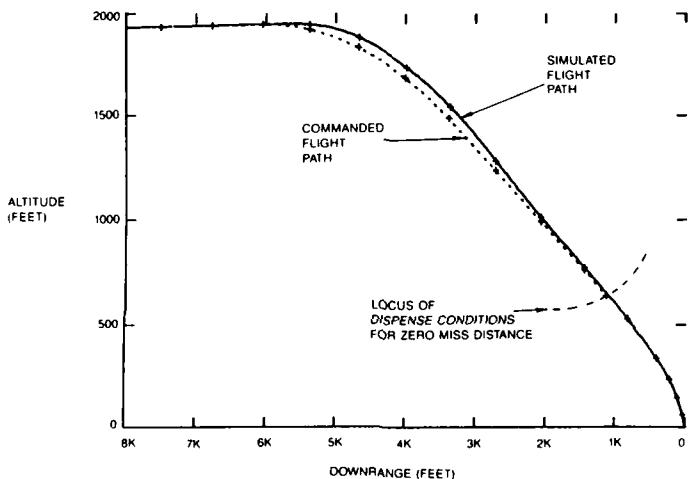
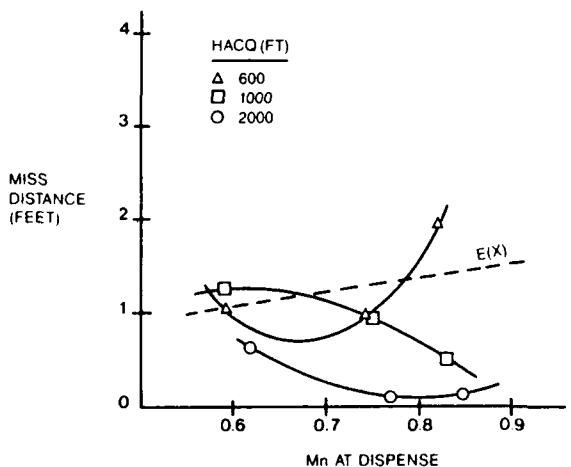


Figure 9. Typical Simulated Trajectory of Weapon System and Submunition

ASSUMPTIONS

- (1) ZERO SYSTEM ERRORS
- (2) 3 DOF SIMULATION (PITCH PLANE)

$$E(X) = \sqrt{\sigma_1^2 + \sigma_2^2}$$

WHERE

$\sigma_1$  = RMS ERROR DUE TO  
BALLISTICS TABLE (0.46 FT)

$\sigma_2$  = RMS ERROR DUE TO  
SIMULATION INTEGRATION  
STEP ( $\Delta t = .005$  SEC)

Figure 10. System Accuracy Without Errors

Table II. Major Error Sources Affecting Terminal Accuracy.

Parameter	Units
<u>Fuzing/Ejection</u>	
(1) error in ballistics tables	feet
(2) timing variations in BKEP start commands	sec.
(3) canister opening delay variation	sec.
(4) primary airbag delay variation	sec.
(5) ejection velocity variation	percent
<u>BKEP Submunition Errors</u>	
(1) weight	percent
(2) drag	percent
(3) chute deploy command	sec.
(4) chute inflation	sec.
(5) chute stability at rocket burn	degrees
(6) rocket ignition command	sec.
(7) thrust misalignment	degrees
<u>Mine Submunition Errors</u>	
(1) weight	percent
(2) drag	percent
(3) drogue deploy variation	sec.
(4) main chute deploy	sec.
<u>System Components</u>	
(1) radar altimeter (scale factor)	percent
(2) accelerometers (bias)	g's
(3) attitude gyro (bias & drift)	deg., deg/sec.
(4) seeker (LOS rate, gimbal angle)	deg/sec., deg.
(5) state estimation (velocity, position)	f/s, feet
<u>External Errors</u>	
(1) atmospheric temperature	degrees
(2) winds	f/s
(3) target altitude (terrain variations)	feet

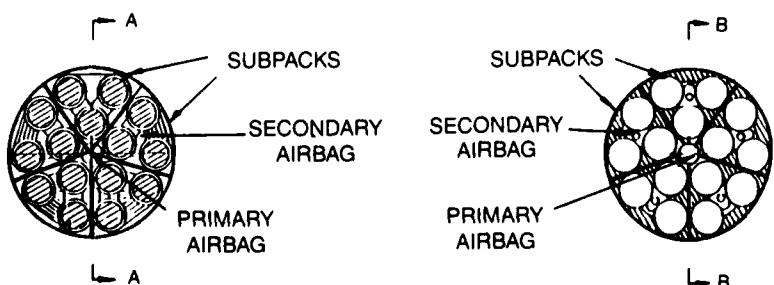
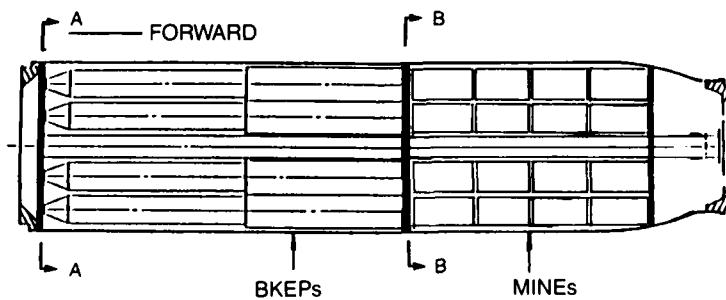


Figure 11. Packaging Concept for BLU-106/B BKEP and HB876 MINES

The errors in the parameters which define the burst condition: altitude (h), speed (v), and dive angle ( $\gamma$ ), also create errors in the downrange distance. As described previously, these parameters or system states are estimated from onboard sensors and the seeker which tracks the target. The sensitivity of downrange distance (X) to dive angle can be defined as the differential: ( $\Delta X/\Delta\gamma$ ).

Table III contains six of the more important sensitivities for both BKEP's and Mines about a nominal dispense point. Notice the difference in sensitivities between the two warheads. For example, the Mines are more than twice as sensitive to winds as the BKEP's. The BKEP's downrange error due to chute errors is more than five times greater than that of the Mines.

A complete accuracy evaluation must include the above sensitivity coefficients as well as the measured system errors and expected environmental variations. Since nonlinearities and interactions between variables are likely, a complete evaluation would employ covariance analysis or Monte Carlo techniques.

Table III. (U) System Sensitivities to Parameter Variations About the Nominal Dispense Condition

Parameter	Sensitivity	Units	(BKEP Value/Mine Value)
dive angle	( $\Delta X/\Delta\gamma$ )	ft/deg	2.5
altitude	( $\Delta X/\Delta h$ )	ft/ft	8.6
speed	( $\Delta X/\Delta v$ )	ft per f/s	1.1
wind	( $\Delta X/\Delta V_w$ )	ft per f/s	0.4
drag	( $\Delta X/\Delta C_D$ )	feet	0.7
chute time	( $\Delta X/\Delta t_c$ )	f/s	5.7
ejection time	( $\Delta X/\Delta t_{ej}$ )	f/s	1.4

## 5.0 SUBMUNITION DISPENSING METHODS

Once the basic elements of the dispenser system are defined, such as the type of submunition to be used, air bag dispensing, etc., analyses must be conducted to precisely define individual pieces of the system design. For example, with the dual airbag system used to expel the submunitions, the magnitude of the total ejection velocity imparted to the submunitions must be determined as well as the relative magnitudes of the primary/secondary ejection velocities. In addition, such system design parameters as the altitude, speed, and flight path angle of the weapon at the time of the dispense event must be defined, as well as a determination of the system's sensitivity to errors in those flight parameters. Many of the design parameters cannot be determined solely on the basis of dispensing and patterning concerns, but must also address the constraints imposed by the terminal guidance algorithms.

This section discusses how the system parameters are selected on the basis of packaging, dispensing, and patterning consideration. Also presented are some reflections on the performance of the system to slight errors in the parameters which will occur in an operational environment.

### 5.1 Dispensing and Packaging Concepts

The packaging design of the system relates to how the submunitions and dispensing mechanisms (air bags, hot gas generators, dunnage, etc.) are placed into the SUU-54 dispenser weapon to maximize the payload of BKEP and MINE submunitions. Among the first considerations are whether the submunitions will be dispensed radially away from the dispenser longitudinal axis or laterally outward (to the sides of the dispenser). Previous sections of this report have indicated that the selected system uses radial ejection. This selection was based on trade off studies of the two approaches based on cost, complexity of the dispenser design, technical risk, and effectiveness of the resultant system. These trade off studies will not be presented herein, but the general results are that the radial technique is better suited to the SUU-54 dispenser weapon, allows for more BKEP and MINE submunitions, and allows better pattern control.

At the initiation of the dispense signal from the dispensing algorithm, a series of cutting charges are enabled, which cut through the metal skin of the dispenser weapon. After a short delay to allow the covers to clear, the submunitions are ejected. The dispense system uses dual airbag ejection in which a hot gas generator, initiated by a pyrotechnic charge, is vented through a central mandrel to the primary airbags. The sudden pressurization of the primary airbags results in the ejection of subpacks containing the submunitions. The subpacks consist of three submunitions each, encased in rigid foam dunnage to form triangular shaped subpacks. Figure 11 shows the packaging concept for the BKEP and MINE submunitions. Because of their length, only one row, or layer, of BKEP submunitions can be placed in the SUU-54 dispenser. The shorter MINE submunitions allow for four layers to be placed in the dispenser, with the same cross section packaging of five subpacks of three submunitions each. This packaging concept allows for a total of fifteen BKEP and sixty MINE submunitions. Although not presented herein, trade off studies were performed for alternate methods of arranging the submunitions within the dispenser, but the one presented has the most flexibility in terms of pattern control.

One of the first things which must be defined before the dispensing and pattern analysis can begin is the definition of the required pattern. The pattern dimensions are a function of the expected accuracy of the system and operational effectiveness considerations. The following example is intended to show the ramifications of pattern size on the dispense system design. Figure 12 is an illustration of various patterns superimposed on a typical aircraft runway target of 200 foot width. Assuming that the minimum "clear space" between impact craters which would allow an aircraft to taxi through is 40 feet, one sees that a system with no errors and perfect accuracy could have a relatively small pattern and achieve a high probability of cutting the runway, independent of the direction of approach of the weapon. However, for the design of a real system with errors and imperfect accuracy, the problem becomes more complex. For a 90 degree approach (across the runway), a long pattern is required to accommodate the longitudinal errors and still provide a runway cut. Pattern width could be relatively small. The required pattern length should increase (over the no error pattern size requirement) by approximately twice the longitudinal error. Also, since the pattern length exceeds the runway size, less than a full load of submunitions will impact the runway, with some loss of effectiveness. For a 0 degree approach (down the runway), pattern length can be relatively small, but the width must be sufficient to cover the runway width plus approximately twice the lateral error. While pattern length can be obtained easily by adjusting the submunitions' parachute timing, pattern width must be generated by the ejection mechanism. Considering an intermediate approach angle such as 45 degrees, the length and width chosen on the basis of the 0 and 90 degree approach scenarios could result in gaps in the impact pattern which would dictate a slightly longer pattern to fully cover the runway and provide the necessary cut. With these considerations in mind, the selected pattern is an ellipse with a length of 300 feet and a width of 200 feet. This example clearly shows the relationship of pattern size requirements and system accuracy, and therefore, system cost analyses must be critically weighed relative to accuracy improvements.

One area which may not be obvious is why the system is not designed for a fixed approach angle to the runway instead of designing it for an arbitrary angle. The reason is that if the system is designed for one approach angle, the dispenser weapon must fly a specified heading to intersect the runway at that angle. That places constraints on the heading of the launch aircraft, which may subject it to attack by airport defensive placements. From an aircraft survivability standpoint, and for greater mission flexibility, it is desirable to be able to approach the runway from any angle and still have a high probability of runway cut. It is possible that the overall probability of cut for such an "all azimuth attack" system at a specified approach angle will be lower than that for a system which is optimized for that angle, but the overall effectiveness and usefulness of the all azimuth attack weapon system will be greater.

EXAMPLE ASSUME 200 FT RUNWAY 40 FT CLEAR SPACE  $E_H = 75$   $E_L = 375$  FT

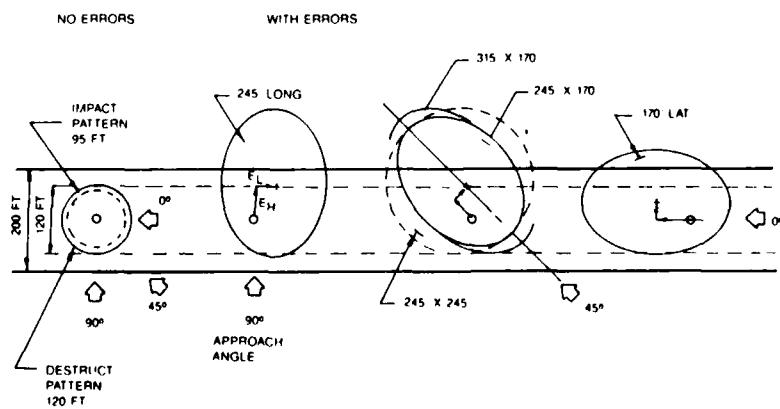


Figure 12. Pattern Size Criteria

Once the basic pattern shape and size requirements are defined, the dispenser system design must be chosen to meet those requirements. One of the first things to be defined is the ejection velocity to be imparted to the submunitions by the dispenser system. The pattern width is a function of the total ejection velocity (sum of the primary ejection velocity of the subpack from the dispenser and the secondary ejection velocity of the individual submunitions from the subpack) and the submunition parachute deployment delay. The maximum total ejection velocity using current airbag technology is approximately 90 feet per second. The parachute deployment delay is determined by the sometimes conflicting requirements of the terminal guidance algorithms and the pattern requirements and by limits established by the submunition design. For brevity, the interrelationship between these two areas will not be presented herein, but in summary, a parachute deployment delay of 0.5 seconds is adequate to meet the requirements of the present system. Once the parachute deployment delay is established, the selection of the total ejection velocity is proportional to the pattern width. For the required width of 200' feet, the total ejection velocity requirement is approximately 80 feet per second. The allocation of the total ejection velocity into primary and secondary ejection velocities must be addressed next. Because of space limitations in the small subpacks, there is not room for large gas generators and airbags, so the secondary ejection velocity must be smaller than the primary ejection velocity. Fortunately, from a pattern uniformity standpoint, the optimum ratio of primary-to-secondary ejection velocities is approximately two-to-one. Figure 13 illustrates the effect on the pattern of different primary/secondary ejection velocities. From the figure, it is apparent that for a much larger primary ejection velocity, the submunitions from each subpack are well separated, but those from a single subpack are too close together, indicating that the secondary ejection velocity is insufficient to spread the subpacks' submunitions to achieve a uniform pattern. For a secondary ejection velocity which is approximately equal to the primary ejection velocity, the secondary ejection spreads the subpacks' submunitions too much, causing a "torus" shaped pattern. The ratio of two-to-one provides a uniform pattern.

The flight condition of the weapon at the initiation of the dispense event must also be specified. The weapon incorporates an altitude hold system using a radar altimeter to maintain the specified altitude. The range of allowable altitudes for the weapon system is from 200 feet to 2000 feet, in increments of 200 feet. The altitude is chosen by the WSO through the data link control. The altitude is usually chosen low to prevent countermeasures from being directed against the weapon by defensive forces. The altitudes most examined in the dispensing analysis are 1000 feet and 400 feet. The patterns which result from the two altitudes do not differ appreciably. As mentioned previously, the pattern width is a function only of the parachute deployment time delay and the total ejection velocity. The pattern length is slightly longer for the higher altitude at dispense because the terminal trajectory of the BKEP after release of the parachute is not vertical, but is typically approximately 35 degrees from the vertical. The downrange distance of the centroid of the pattern is the major difference between the two cases. Because of the higher altitude at dispense, the pattern for the 1000 foot dispense is about 200 feet farther downrange. However, one should note that the altitude at the time of the dispense is determined by the dispensing algorithm, not the midcourse altitude selected by the WSO, so that the actual variations in dispense altitude will be much smaller than the range of altitudes just discussed. The range of altitudes that the dispense algorithm uses is in the range of 300 to 600 feet. The weapon speed at the time of dispense is expected to be in the range of 0.6 to 0.8 Mach. Studies have indicated that the patterns are not appreciably affected by the dispense speed, except for the downrange distance of the pattern centroid. The dispense algorithm will be aware of an estimate of the speed from the EKF so that adjustments in the time of dispense can be made to further reduce the speed effects. The last major parameters of the weapon's flight condition at the initiation of the dispense event are its flight path angle and angle of attack. One would like for the angle of attack to be zero to prevent the dispenser covers from being held in place by aerodynamic forces, which might cause interference with the ejection of the submunitions. Since the weapon is flying approximately a ballistic flight path, the angle of attack should be small. The flight path angle at the time of dispense is under the control of the digital controller, or autopilot, and is generally in the range of 0 to 20 degrees below the horizontal.

## 5.2 Submunition Patterns

Most of the discussion so far has dealt with the patterns of the BKEP submunitions. However, one of the anticipated problems was in being able to overlay the patterns of the BKEP and MINE submunitions, since they have such different ballistic characteristics and only the parachute deployment delay times are available for changing the resultant patterns. Figure 14 shows the MINE patterns which are developed with the two extremes of parachute delay times which are available. The "baseline" BKEP pattern, which is the goal for the MINE pattern, is shown in Figure 15. The fifteen points represent only one of the four rows of MINEs which are contained in the dispenser, and all were set with the same parachute delay, although each submunition can have a different delay time, if desired. (The delay time is not selectable in flight; rather, it is a mechanical switch set when the submunitions are loaded into the subpacks prior to being loaded into the canister. The precise value for the delay

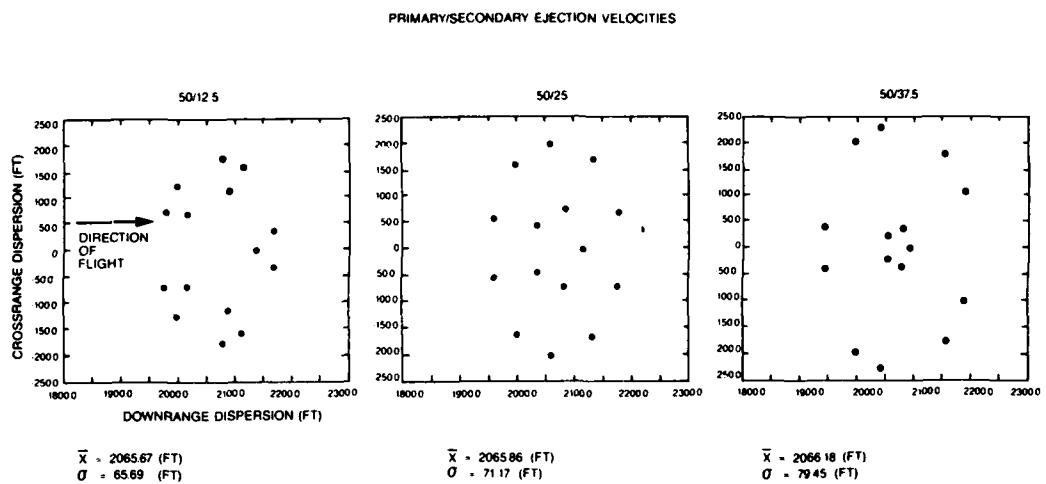


Figure 13. BKEP Patterns as a Function of Primary/Secondary Ejection Velocity Ratios

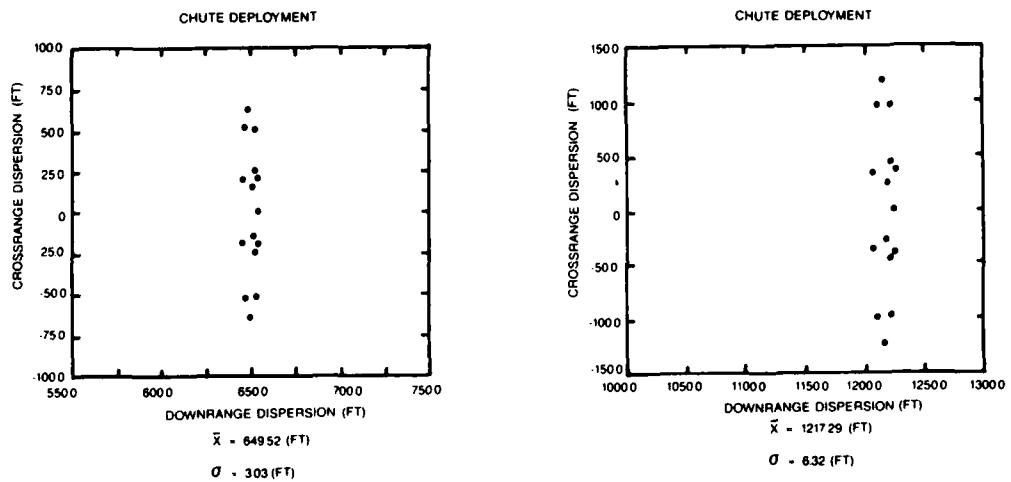


Figure 14. MINES Pattern as a Function of Chute Deployment Initiation Time

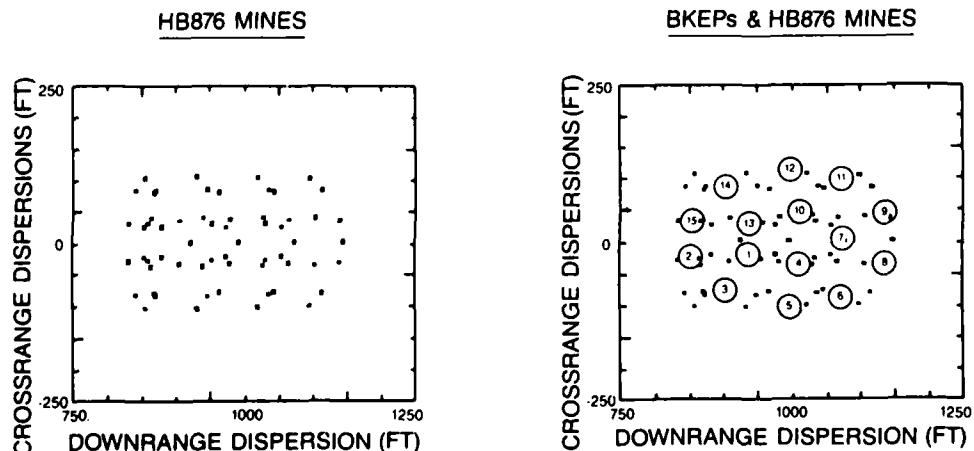


Figure 15. BKEP/MINE Baseline Ground Impact Pattern

must therefore be chosen to be optimal over all expected flight conditions). Figure 14 shows the large variation of downrange distance that is available solely through variations in the parachute delay. Also note that the width of the MINE pattern is function of the parachute delay as was the case with the BKEP patterns.

Since the dispenser contains four layers of MINE submunitions along its axis, each layer can be ejected with its own primary airbag to allow more flexibility in the pattern. This feature, along with parachute deployment time delays, is used in the final design to allow the required overlap of the BKEP and MINE pattern, as shown in Figure 15. Note that in the pattern showing only the MINE impacts, the four layers of submunitions are clearly visible. The resultant BKEP/MINE pattern shows an excellent overlap of the two different types of submunitions and that the desired 280 by 210 pattern size has been obtained.

#### 6.0 CONCLUSIONS

It is feasible to modify a tactical standoff weapon with a unitary warhead to be able to deliver dispenser-type warheads against area targets. A guidance algorithm has been developed which provides high accuracy with autopilot grade sensors, a radar altimeter, and terminal seeker. The concept does not require an inertial navigation system with sophisticated alignment procedure. The concept is easily adapted to various submunitions and is relatively independent of delivery vehicle characteristics. The algorithms are easily implemented with state-of-the-art digital processors.

The generation of accurate and repeatable submunition impact patterns is made possible by a dual airbag ejection system. The force/stroke characteristics can be tailored to a wide range of payloads and pattern diameters. Pattern uniformity is assured by a secondary ejection of subpacks. The airbag technology required is well within the state-of-the-art and has only a minor impact on total system weight.

#### REFERENCES

- (1) AGM-130 Preliminary Engineering Evaluation Program Final System Design Analysis Report (U), Contract F08635-84-C-0255, Rockwell International - Missile Systems Division, 10 September 1984.
- (2) AGM-130B Preliminary Engineering Evaluation Program Systems Analysis Report (U), Contract F08635-84-C-0255, Rockwell International - Missile Systems Division, 13 May 1985.

HIGH ALTITUDE, LONG ENDURANCE RPV  
DESIGN TECHNOLOGY STUDY

by

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SUMMARY

The Naval Air Development Center has been conducting a design technology study of high altitude, long endurance Remotely Piloted Vehicles, RPV's, for Navy possible mission applications which might include surveillance, over the horizon communications and targeting, among others. Phase I of the study was to investigate technology levels and potential technology breakthroughs that can provide vehicle endurance of >100 hours at altitudes >60K feet, and to incorporate the technologies into conceptual vehicle designs. The results of the Phase I study are presented.

INTRODUCTION

Performance parameters addressed in the RPV design technology study are for long endurance (>100 hours) and high altitude (>60K feet). A possible mission application for the long endurance RPV is surveillance, thus the conceptual vehicle designs incorporate phased array radar - which has been under development at NAVAIRDEVCECEN for sometime - and other sensors including infra-red search and track, IRST, and electronic support measures, ESM.

A parallel mission study was conducted which investigated items such as basing, radius of action, vehicle speeds, number of vehicles deployed, time on station, etc. Vehicle performance objectives were thus applied in the development of the vehicle designs. The operational concept is to loiter at very high altitude, conduct wide area surveillance of potential enemy threats and to communicate information back to the tactical fleet commander. The total vehicle surveillance and communications range can be extended by communications relay to either existing surveillance aircraft or to other RPV's which are situated to provide line of sight communication relay. A number of alternate deployment arrangements have been investigated.

Implications of vehicle design requirements to achieve long endurance at high altitude are indicated by the Brequet equation for propeller driven vehicles in Figure 1. High values of the endurance parameter  $C_L^{3/2}/C_D$  are needed; high aerodynamic L/D to minimize thrust required and high  $C_L$  (or low speed) to reduce engine power requirements. First order effects of low engine specific fuel consumption, BSFC, and high propeller efficiency  $\eta_p$  are also noted. The other driving parameters for vehicle endurance are low values of wing loading, W/S, and low empty weight fraction. The latter term actually embodies three fundamental design parameters which are low structural weight fraction, low engine system weight and of course a high fuel to gross weight fraction.

Several alternative design concepts were evaluated, ranging from a conventional wing/fuselage/canard vehicle that houses the radar in the fuselage, to a flying wing configuration that incorporates the radar within a thick suction (BLC) wing profile.

Applicable engine/propeller propulsion systems were investigated to meet the high altitude, low BSFC requirements. Recommendations and performance estimates are given for a near term propulsion system - applicable to a demonstrator - as well as for a more far term system for an operational vehicle.

Structural concepts investigated employed advanced composite materials in arrangements to exploit thick wing design and achieve minimum structural weight.

The Phase I study has shown that necessary combinations of design technologies are available or are projected to be available to meet the mission objectives. The selected vehicle conceptual design is a flying wing with sufficient thickness in a passive (no BLC) section to house the radar in an orientation that permits a full 360° of radar scan.

The Phase I results provide the basis for development plans for Phase II which are to conduct critical component tests and analysis leading to development of a proof of concept demonstrator vehicle.

CONFIGURATION ALTERNATIVES

At the beginning of the design study, analyses were conducted to determine the general level of technology that was required, as well as what was projected to be available, to achieve high altitude RPV endurance exceeding 100 hours at altitudes exceeding 60K feet. Fuel fraction requirements were determined from the Brequet endurance equation, with rough allowances made for climb and transit.

$$\Delta W_f/W_1 = 1 - \frac{1}{(Rt+1)^2}$$

$$\text{where } B = \frac{\text{BSFC}}{37.9} \quad \frac{\rho_0}{\rho} \quad \frac{W_1}{S} \quad \frac{1}{C_L^{3/2}/C_D}$$

$t$  = endurance in hours  
 $W_1$  = vehicle weight at begin endurance  
 $C_L^{3/2}/C_D$  = aerodynamic endurance parameter  
 $\rho_0/\rho$  = sea level to flight altitude density ratio  
 $\Delta W_f$  = fuel used

The weight fraction equation,

$$\frac{W_F}{W_G} + \frac{W_{PAY}}{W_G} + \frac{W_{ENG}}{W_G} + \frac{W_{ST}}{W_G} + \frac{W_{FIXED}}{W_G} = 1.0$$

was solved to estimate design gross weight,  $W_G$ , as functions of the driving parameters in the endurance fuel equation, i.e., engine/propeller efficiency, flight altitude, wing loading and aerodynamic efficiency over a range of structure to gross weight ratios. Preliminary indications were that turbocharged engine BSFC values could approach 0.30 to 0.35, and that propeller efficiencies of nearly 0.90 and specific power values in the range  $W_E/HP = 3$  to 5, could be achieved. An estimated range of payload weights for expected mission applications was 1000 to 2000 pounds. The weight fraction required for fixed equipment was estimated to be 0.10 based on earlier design studies. One typical result of the design sensitivity analysis is shown in Figure 2 which is for a mission endurance of 122 hours at 70,000 feet. Vehicle gross weight varies strongly with wing loading as shown. The effect of structural weight fractions of .16, .18, .20 and aerodynamic parameters of 30 and 35 were determined at fixed values of payload and engine parameters as listed on the figure. These preliminary sensitivity results indicated the strong need for low wing loadings, <8, as well as the need for aerodynamic efficiency  $C_L^{3/2}/C_D > 30$  and for structural weight fractions <0.20 in order to obtain a design solution for this altitude and endurance. Comparisons of low drag wing sections at an aspect ratio of  $AR = 20$  and  $R_N = 1.0 \times 10^6$  were made as shown in Figure 3. Indications were that with careful design, an endurance parameter of  $C_L^{3/2}/C_D = 30$  could be achieved. Based on these results, and initial estimates of achievable goals for engine performance, structural weight fraction and aerodynamic efficiency, design gross weights of 8,500 pounds (1,000 pound payload) and 17,000 pounds (2,000 pound payload) were selected for preliminary configuration layouts. Table 1 lists the candidate configurations that were considered at the early stage of the design study. Configuration 1, a wing/fuselage/canard configuration, had been studied earlier for shipboard operation. This design arrangement housed the HARPSS radar in the fuselage, giving radar coverage of 240° - less than desired. The 1,000 pound payload weight assumed radar to be the principal payload. This configuration two-view is shown in Figure 4 with pertinent geometry characteristics listed.

An alternative arrangement which would house the radar in a thicker ( $t/c = .18$ ) passive wing is shown as configuration 2 in Figure 5. The take-off gross weight of 17,000 pounds (2,000 pound payload) with a wing loading of 6.0 PSF allowed for placement of fore and aft radar arrays in the wing with estimated radar apertures as shown. The aft stabilizer arrangement insures adequate stability and control, although with some penalty for empennage drag. As will be discussed, a major part of the design analysis study was directed to the flying wing configuration, (configuration 3 and 3A). However it is to be noted that much of the design technology studied is also applicable to this aft tail configuration #2, i.e., wing structural analysis, propulsion system, radar size and performance, and wing aerodynamics. It is in the area of aerodynamic drag and performance analysis that further work is required on this configuration. This is of course in addition to stability and control analysis, although this area is expected to be relatively straight forward. It is felt, therefore, that this configuration is a reasonable backup approach to the flying wing, should further analysis indicate this desirability.

An advantage was initially seen for a high thickness ratio suction (BLC) wing because a deep aperture of the radar could be installed within the wing. The efficiency of the 50% suction wing is shown in Figure 3 for zero internal loss. The 30% suction wing result was obtained from British test data and was not considered to be optimized. The success of this type of wing profile hinges on the degree to which unseparated flow is obtained with suction power levels - converted to equivalent drag power ( $C_{D_0}^{-1}$ ) - comparable to the passive wings that are shown.

A third configuration approach then was the flying wing configuration which is shown in Figure 6.

#### CONFIGURATION #3 - SUCTION WING ANALYSIS

In addition to the stability and control and wing structural analysis that were carried out for the thick suction wing, configuration #3, a major area of concern was the requirements for suction power and the resulting equivalent drag power of this vehicle. Efforts were therefore directed to an evaluation of the wing pressure distribution, and boundary layer thickness at the high altitude low Reynolds number flight conditions and the resulting suction flow control system that would be required. A laminar boundary layer displacement thickness of 0.125 inches was calculated at the location of the suction slot. It was assumed that about 70% of this displacement thickness mass flow should be removed to maintain flow attachment and low external drag in the aft portion of the airfoil. This quantity of mass flow is approximately 25% of the total boundary layer mass flow. For suction power analysis, it was assumed that the full wing span would require suction (top and bottom) for the 50%  $t/c$  wing.

A detailed suction system analysis was carried out. The essential results of that analysis is that the suction system for this vehicle requires 316 horsepower to remove the boundary layer mass flow, which is about three times the amount inferred from data in the British reports - although it is fair to say that internal losses and suction power requirements were not investigated in detail in the British work. It was learned that one major reason for the high horsepower requirement is a consequence of the high altitude flight condition. The flow ambient pressure condition is such that with allowance for in-

ternal pressure losses, a large pressure-ratio compression is required to bring the flow to exhaust pressure conditions which further increases the power requirement.

The results of this suction system analysis when converted into the propulsion system power required for endurance flight indicated that the suction wing was not a good choice for overall aerodynamic performance. Table 2 provides a summary of the effective zero lift drag,  $C_{D_0}^1$ , coefficients of the suction wing and the consequent values of endurance parameter  $C_L^{3/2}/C_D$  for the vehicle. Note that the values of  $C_{D_0}^1$  which are due to internal suction power are the external drag equivalent, i.e., resulting in an equivalent propulsive power to maintain level flight, as that of a passive wing. A design goal  $C_{D_0}^1$  of .016 which was consistent with the  $C_L^{3/2}/C_D$  required for the 120 hours of endurance was considered achievable based on the British test results, as shown in Table 2. However, the consequence of the high suction power causes an effective  $C_{D_0}^1$  of .045 which gives a  $C_L^{3/2}/C_D = 17$ . This is not competitive with much higher performance passive wing sections.

#### CONFIGURATION 3A - PASSIVE WING POINT DESIGN

##### Configuration Description

A 17,000 pound gross weight flying wing configuration having an 18% passive airfoil (Wortmann section FX61-184) is the basis of the presently selected point design for the high altitude RPV. The configuration layout is presented in Figure 7. Stability and control considerations led to the selection of a taper ratio of 0.2 with the elevons and rudder sizes and locations as shown on the drawing. The constant radar height of 18" over a 30 foot span is consistent with performance requirements when a modified radar dipole length is accounted for. Twin pusher, puller engines with 20 foot diameter propellers are used. This arrangement results in a main gear length (static compression) of approximately 8 feet. The engine nacelle is mounted semi-submerged on the upper wing for maximum ground clearance. Nose gear length will be such as to give a static ground angle of 0°. A -3° linear twist is tentatively specified which in combination with negative effective twist due to wing bending will produce a positive  $C_{M_0}$  to minimize trim drag. The wing leading edge sweep is 30° and the trailing edge sweep is 23°. The wing aspect ratio selected is 20. Accounting for the canted wing tip fins, the effective aspect ratio is estimated to be AR eff. = 21. The wing tip fins provide directional stability and also act as all moveable surfaces for directional control during take-off and landing. A 25% chord rudder is provided for cruise flight directional control and is geared to the main vertical to act as a flapped vertical surface when used as all moveable. The elevons are simple hinge surfaces to provide both pitch and roll control throughout all flight phases.

A summary of weights, payload and design point characteristics for configuration 3A are presented in Table 3. A schematic of Radar Coverage with the fore and aft wing arrays is shown in Figure 8.

##### Performance

A design point mission profile with assumed nominal engine performance is presented in Figure 9. A minimum time to climb of 3.6 hours is calculated. The cruise out duration of 6.2 hours is computed for best range at 70K feet. Maximum speeds and minimum time occur at the higher altitudes and since maximum specific range is not greatly different from maximum speed, the minimum fuel profile is also close to a minimum time to reach deployment range. The return leg duration is much longer due to the lower weight and is not near max speed. A two hour reserve fuel allowance is provided at the end of the mission. A mission endurance of 120 hours is provided under these conditions with the fuel quantity of 9540 pounds as shown.

The sensitivity of the vehicle mission endurance to variations in engine performance and flight altitude is presented in Figure 10. The substantial endurance gain at lower altitudes to 60K feet are evident and as such make up for sizeable deficits in engine performance or hot day conditions.

Wind speed versus altitude profiles were determined using NASA wing survey data at higher altitudes as shown in Figure 11. A representative RPV climb profile for a  $C_L = .70$  is superimposed. For altitudes above 40K, vehicle speeds become substantially greater than the 1% wind profile, i.e., 300'/sec cruise out speed and 272'/sec to 200'/sec loiter speeds.

##### Stability and Control

Longitudinal and lateral/directional stability and control characteristics for the hi-flyer flying wing concept were determined using preliminary design type of analysis methods. The static longitudinal, lateral and directional stability characteristics were determined in addition to the trim requirements for cruise, take-off and landing flight conditions. An initial assumption of a rigid airframe during the analysis was followed by a similar analysis which included preliminary estimates of aeroelastic effects.

Aerodynamic control for the configuration is provided by elevons in the pitch and lateral axes and by all moveable winglets in the directional axis. Both the elevons and the winglets were assumed to be symmetrical airfoil sections (64-018) with sufficient thickness to delay stall until higher deflections. The winglets act as vertical stabilizers, with a rudder whose deflection is geared to the deflection of the winglet at a 1:1 ratio, i.e., each degree of deflection of the winglet results in a degree of deflection of the rudder (relative to the winglet).

The center of gravity and pitch moment of inertia were determined in the takeoff and landing flight conditions assuming a rigid airframe.

The relative locations of the aircraft center of gravity and aerodynamic center result in the stability margins contained in Table 4. The margins are presented for both take-off and landing conditions, the landing margin being significantly higher due to the expenditure of fuel. The stability margins shown for the rigid configuration are quite high but are required to compensate for a preliminary

estimate of a 12% loss in stability due to static aeroelastic effects. This is considered an upper limit on acceptable aeroelastic effects. More recent information indicates that the stability loss may be much less.

The basic aerodynamic characteristics and stability derivatives were determined for the configuration. The negative  $C_m^0$  of the Wortmann airfoil, combined with positive longitudinal stability, results in a condition of higher than desired elevon deflection and consequent trim drag. To avoid this condition a linear twist of the wing was incorporated to convert the negative  $C_m^0$  to a zero  $C_m^0$ . When analyzing the aircraft assuming a rigid airframe a linear  $-3^\circ$  twist was designed into the wing. As an additional benefit the wing twist will also alleviate a tendency towards tip stall, which aids in maintaining the elevon effectiveness.

The static directional stability is quantified by the yawing moment due to sideslip,  $C_n\beta$ . As indicated by the value in Table 4, the hi-flyer is directionally stable. This derivative was calculated assuming both a rigid and elastic airframe, the relative movements of the aerodynamic center and center of gravity in the aeroelastic analysis resulting in a negligible difference.

#### Propulsion System

An extensive survey of propulsion system types, characteristics and projected performance levels was conducted for the High Flyer Study. The various types investigated together with pertinent characteristics of each for this high altitude mission are summarized in Table 5. From all of these considerations as well as considerations of availability in time periods of interest, the selected power plant for near term applications is the spark ignition reciprocating. For far term applications, i.e., 5 to 10 years hence, the three stage turbo charged rotary engine appears to be a proper choice. These recommended power plant systems with associated cooling systems, etc. are summarized in Table 6.

A standard day normal power level of 313 horsepower is for level flight at the design  $C_L = 1.1$  and speed of 161 knots. The power required for standard rate turns during the endurance phase is 344 horsepower as shown. A comparable hot day power required is 363 horsepower. It is expected that the engine system would be designed for the nominal 360 horsepower level as a normal rated power level. The normal rated power would be flat rated from sea level to 70K, thus providing considerably higher thrust levels during climb. The additional power required for avionics and cooling systems are being considered to be provided by use of the engine bottoming cycle and exhaust gas energy. The bottoming cycle uses engine heat in a freon system to drive a turbine to provide power for the avionics system as shown. The engine exhaust gases which are used to drive the two stages of turbo charging for engine inlet air are also diverted to run a turbine to provide power for the fans and pumps used for cooling the inter cooler. The propeller is a two bladed design with a variable blade pitch. The engine blade pitch angle control system will be active during the climb and descent phases of flight whereas RPM control will be operative during the endurance phase to maintain constant  $C_L$  flight as vehicle weight decreases.

#### STRUCTURAL DESIGN STUDY

The structural design study was performed to assess the feasibility of making a light weight design compatible with the expected mission and other vehicle subsystems. The design goal was to achieve a 17% structural weight fraction i.e., 2890 pounds of structure for the 17,000 pound vehicle.

This study examined the 3A and 3 flying wing configurations. These have airfoils with thickness-to-chord ratios of 18% and 50%, respectively. Primary emphasis was applied to the 18% thick wing and only these results are summarized herein.

The structural configuration and wing section are shown in Figure 12. Bending strength and stiffness are provided by the continuous spars located at the 25 and 52% chord locations. Torsional stiffness and strength are provided by a torque box consisting of the sandwich skins and the spar webs. The wing covering consists of the sandwich skins over the forward 52% of chord and a thin skin over the remaining airfoil section. The covering and ribs maintain the aerodynamic shape and smoothness. Composite materials were selected for light weight and radar transparency. Fiberglass and/or Kevlar are used where radar transparency is required. Graphite epoxy is used elsewhere. A foam core is used in all sandwich construction.

#### Structural Design Criteria

The structural design criteria and considerations used in this study are outlined in Table 7. The principal design requirements are strength, local stiffness and global stiffness. The stiffness requirements were found to be most important for the vehicle weight.

A maximum limit vertical load factor of 3 g and 1.25 ultimate factor of safety were selected for the strength requirements. This covers gust, maneuver, landing and dynamic loads.

Adequate local stiffness is provided for member stability and aerodynamic smoothness. The airfoil's lift and drag characteristics depend on this smoothness.

Adequate global stiffness is provided for static aeroelastic characteristics. Sufficient bending and torsional stiffness exist for aerodynamic performance, stability and control.

#### Structural Analyses

Analyses were performed (a) to define the wing's net loading for flight and landing conditions, (b) to determine aerodynamic surface pressures, (c) to define internal loads for member sizing and design, and (d) to define and assess the wing's local and global stiffness properties.

The surface deflections caused by pressure, wing bending and wing torsion are considered appropriate for the airfoil's lift and drag characteristics.

The wing's deflection and twisting were computed to assess global stiffness. The wing is considered to have adequate static aeroelastic characteristics.

#### AIRCRAFT FIXED EQUIPMENT

Initial sizes, weights, and power requirements were estimated for on-board equipment and payload perceived necessary to fly the RPV's mission profile.

Selected types of payload sensors were radar, infrared detector (IR) and electronic surveillance (ESM) and interfleet communications.

The general layout of all avionics, flight control and sensors considered is shown in Figure 13. In addition to these, the weight, type and support equipment for fuel tanks were considered in the fixed equipment summary.

## • BREGUET FORMULA:

$$E = K \cdot \frac{C_L^{3/2}}{C_D} \cdot \frac{\eta}{BSFC} \sqrt{\frac{\rho}{\rho_0}} \sqrt{\frac{s}{W_I}} \left( \sqrt{\frac{W_I}{W_f}} - 1 \right)$$

- HIGH  $C_L^{3/2} / C_D$
- HIGH OVERALL PROPULSIVE EFFICIENCY
- LOW WING LOADING
- LOW EMPTY WEIGHT FRACTION

FIGURE 1  
IMPLICATIONS OF VEHICLE REQUIREMENTS

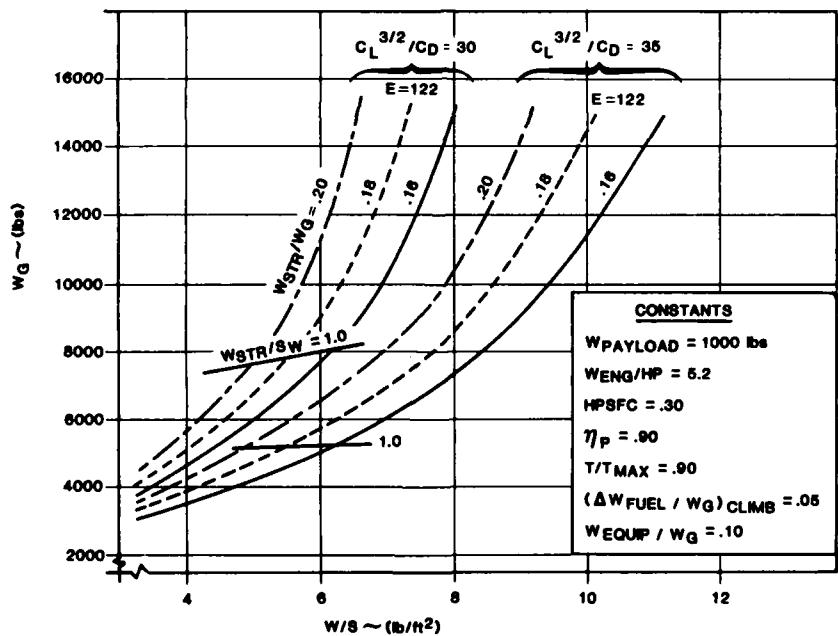


FIGURE 2  
HIGH FLYER DESIGN SENSITIVITIES  
E=122 HRS

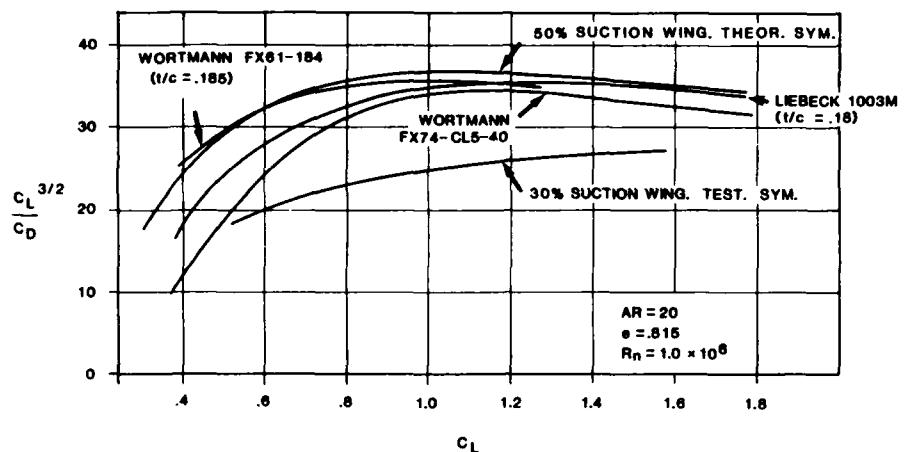
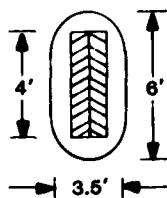


FIGURE 3  
WING COMPARISONS: ENDURANCE FACTOR

	TYPE	TOGW	PAYOUT	WING	REMARKS
1	CONVENTIONAL CANARD	8500	1000	THIN	*TWO CASES LOW & 30 SWEEP *ANTENNAS IN FUSELAGE
2	CONVENTIONAL AFT TAIL	17000	2000	THIN	*30 SWEEP *ANTENNAS IN WING
2A	CONVENTIONAL AFT TAIL	17000	2000	THICK t/c = .50	*30 SWEEP *ANTENNAS IN WING
3	FLYING WING	17000	2000	THICK t/c = .50	SUCTION BLC
3A	FLYING WING	17000	2000	t/c = .18	PASSIVE BLC
4	FLYING WING	17000	2000	t/c = .18	PASSIVE BLC

TABLE 1  
HIGH FLYER CANDIDATE CONFIGURATIONS

WING AREA - 1440 FT  
WING SPAN - 180 FT  
ASPECT RATIO - 17.8  
ROOT CHORD - 14 FT  
TIP CHORD - 4 FT  
TOGW - 8500 LB  
W/S - 5.9  
AIRFOIL WORTMANN  
FX63-137 t/c = .137



SECTION A - A

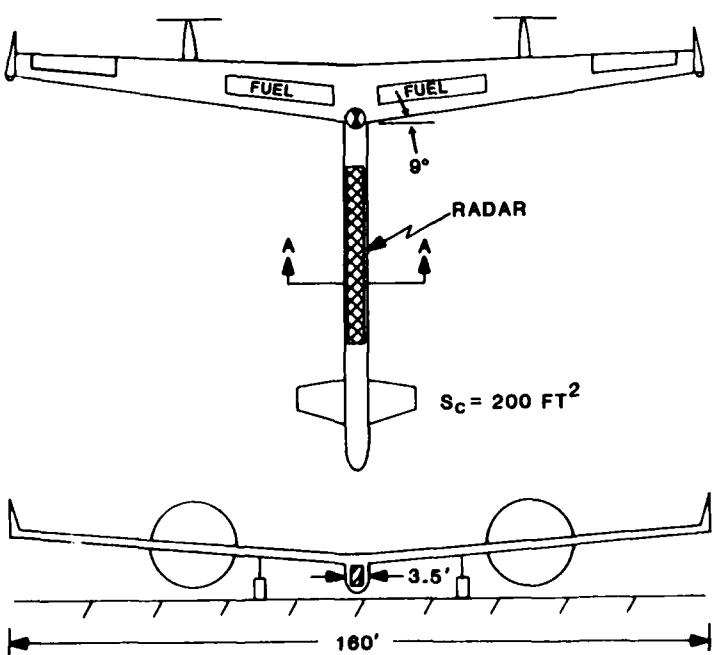


FIGURE 4  
HIGH FLYER CONFIGURATION 1 CONVENTIONAL BASELINE (THIN WING)

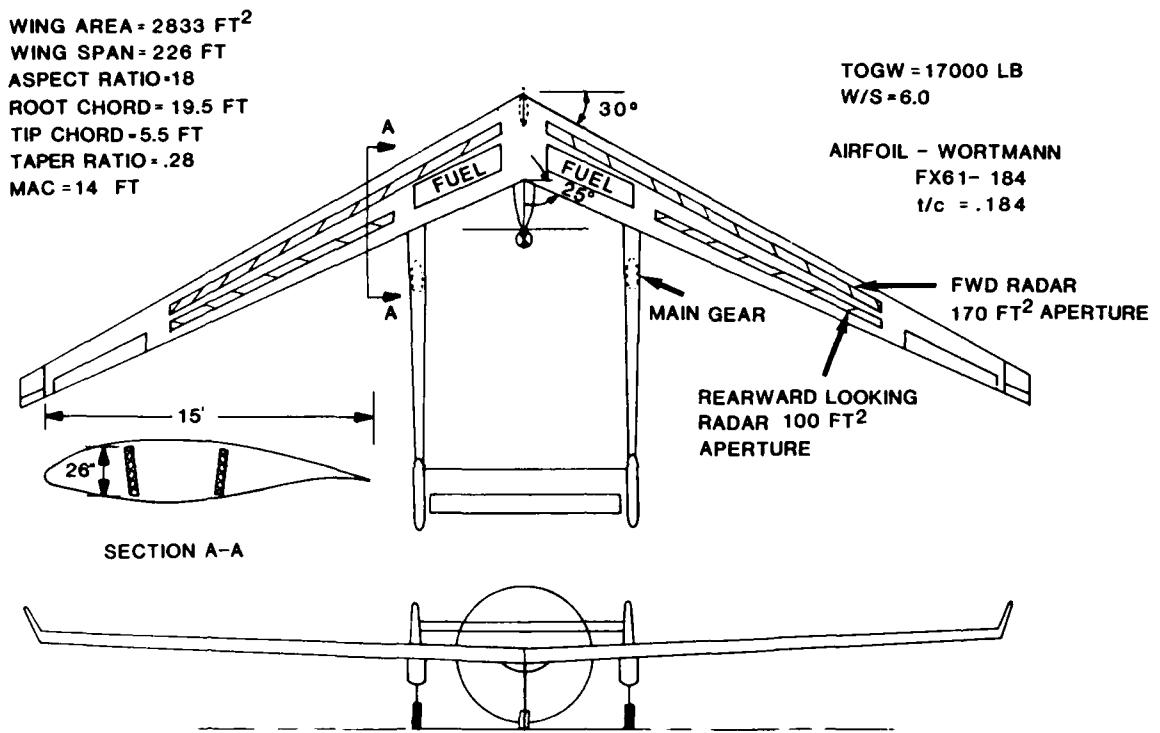


FIGURE 5  
 HIGH FLYER CONFIGURATION 2

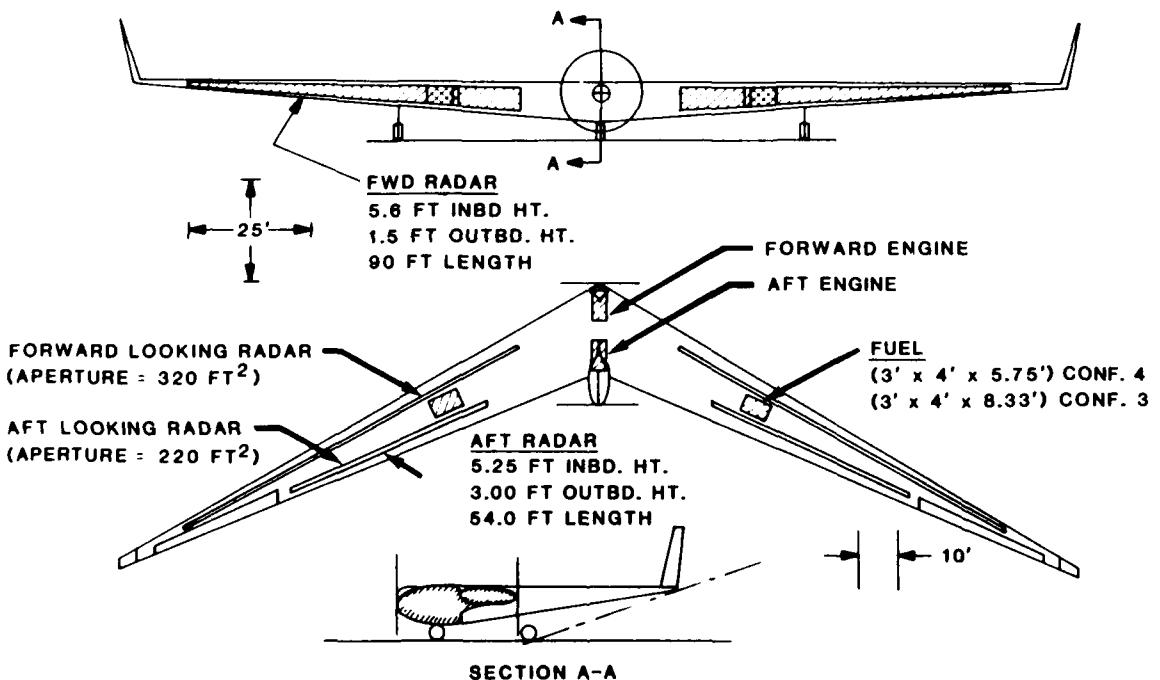


FIGURE 6  
 CONFIGURATION 3 50% THICK WING

CASE	$C'_D_0$	$C_L^{3/2}/C_D$	SUCTION PWR REQ'D AT 70K	$\Delta P_f/q^*$
DESIGN GOAL	.016	30	100 hp	10
BRITISH TEST (30%)	.022	26	142 hp	15
BRITISH THEOR. (30%)	.010		50 hp	4
ANALYSIS	.045	18	316 hp	23 (34)

BASED ON  $RN = 1 \times 10^6$   
LAMINAR B.L. TO SLOT  
 $C_L = 1.4$   $C_L = 1.2$   
DOUG B.L. EST.  $\delta^* = .125^*$   
70% REMOVED UPPER & LOWER  
SURFACE - FULL SPAN  
 $C_D$  WAKE = .0027  
RAKE

$$*HP_S = \frac{q \left( \frac{\Delta P_f}{a} \right) q}{660 \eta}$$

$$(\Delta C_D q_S) V_\infty = 550 HP_S \eta$$

TABLE 2  
HIGH FLYER SUCTION WING

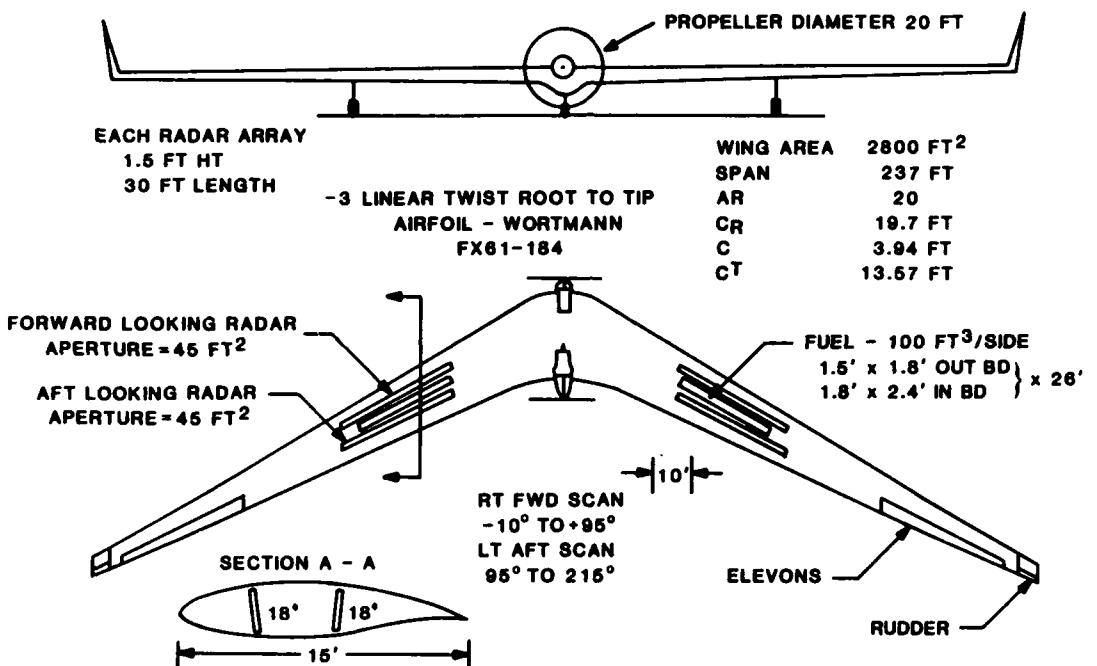


FIGURE 7  
CONFIGURATION 3A 18% PASSIVE WING SELECTED POINT DESIGN

WEIGHTS

- STRUCTURE 2890 LB
- PAYLOAD 1680 LB
- FIXED EQ. 1300 LB
- PROPULSION 1690 LB
- FUEL 9540 LB

WT = 17000 LB

MISSION

- 120 HR8 ENDURANCE AT 70K
- $V = 161$  KTS
- $V_{END} = 118$  KTS

AERODYNAMICS

- $C_L^{3/2}/C_D = 31.2$  @  $C_L = 1.1$
- $(L/D)_{MAX} = 30.6$
- $C_D = .0145 + .0188 C_L^2$

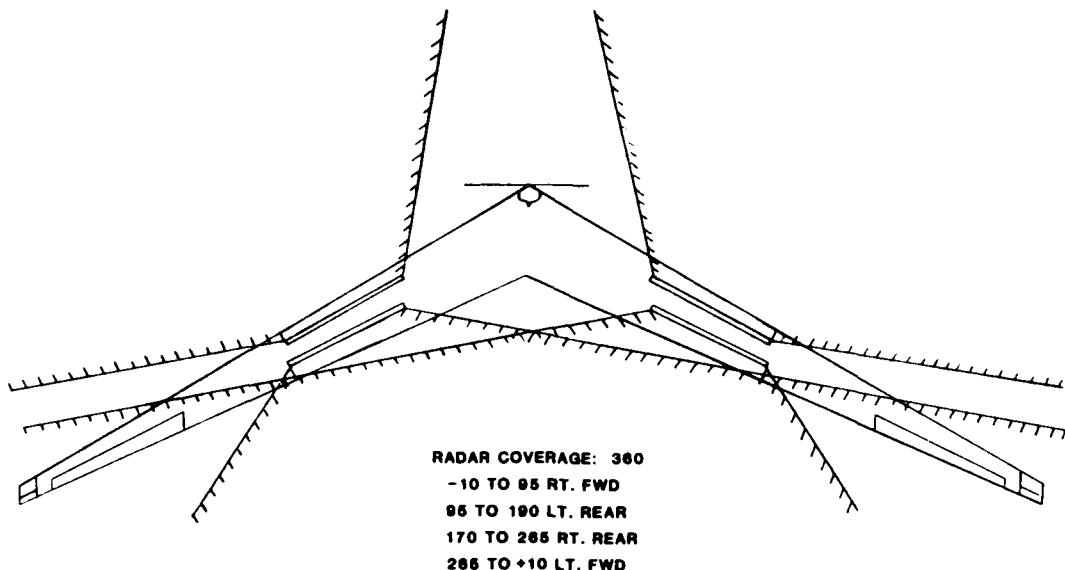
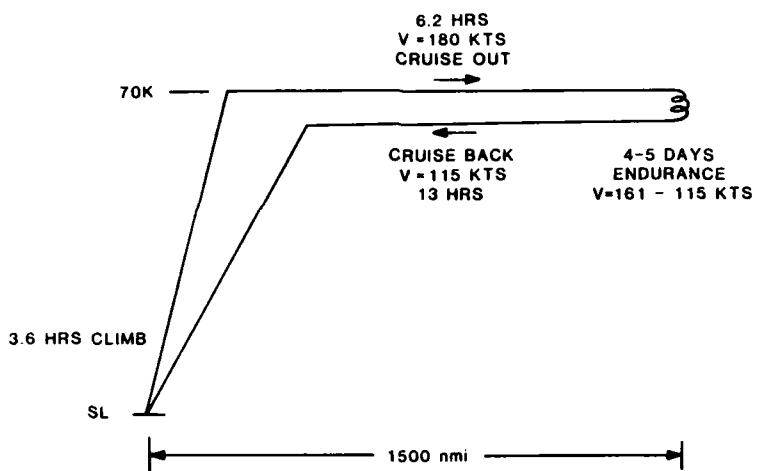
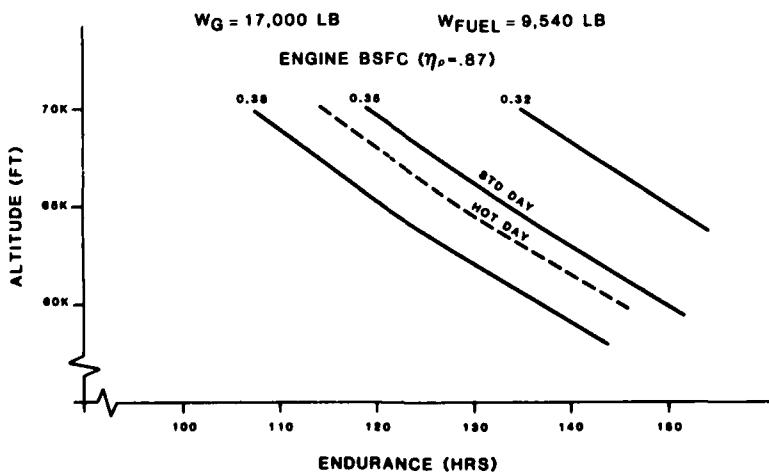
PROPELLUTION

- 360 HP NORMAL RATED AT 70K
- 480 HP MAX AT 70K
- LIQUID COOLED ROTARY
- SFC = .35

PAYOUT

- 1000 LB RADAR
- 330 LB IRST
- 94 LB ESM
- 40 LB COMM. RELAY
- 16 LB DATA PROCESSOR
- 100 LB EXTRA ALLOWABLE

TABLE 3  
CONFIGURATION 3A CHARACTERISTICS SUMMARY

FIGURE 8  
RADAR COVERAGEFIGURE 9  
SURVEILLANCE MISSION PROFILEFIGURE 10  
HIGH FLYER CONFIGURATION 3A

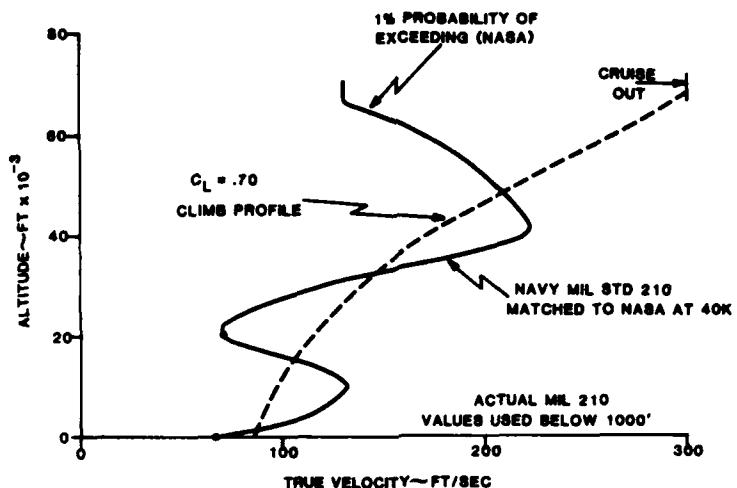


FIGURE 11  
MAXIMUM STEADY WINDS DURING CLIMB

• LONGITUDINALLY STABLE

	<u>RIGID</u>	<u>AEROELASTIC</u>
TAKEOFF CG - FS 30.4	12% MAC	6%
LANDING CG - FS 29.6	18%	12%

TRIM REQUIREMENTS

BEGIN CRUISE	$\alpha = 8^\circ, \delta = -3^\circ$	$\alpha = 8^\circ, \delta = -1^\circ$
END CRUISE	$\alpha = 8.5^\circ, \delta = -5.5^\circ$	$\alpha = 8.2^\circ, \delta = -3^\circ$

- DIRECTIONALLY STABLE -  $C_{n\beta} = .0012/\text{deg}$
- DIRECTIONAL CONTROL ADEQUATE FOR AT LEAST 25 Kts CROSSWIND
- LATERAL CONTROL ADEQUATE FOR AT LEAST 30 Kts CROSSWIND

TABLE 4  
STABILITY AND CONTROL

<u>PARAMETER</u>	<u>ENGINE</u>			
	<u>RECIPROCATING SPARK IGNITION</u>	<u>TURBOPROP</u>	<u>DIESEL</u>	<u>ROTARY</u>
TURBOCHARGED (PR)	YES (30:1)	NO	YES (64:1)	YES (45:1)
STRATIFIED CHARGE	YES	NO	YES	YES
MIN BSFC (LBS/HR/BHP)	0.36	0.33-0.36	0.30	0.32
SPECIFIC WEIGHT <sup>(1)</sup> (LBF/BHP)	2.7	1.2	2.7	2.1
FUEL TOLERANCE	POOR REQ'S AVGAS	POOR	MULTI FUEL NOT AS GOOD AS ROTARY	MULTI FUEL
MAJOR ADVANTAGES	<ul style="list-style-type: none"> <li>• TECHNOLOGY WITHIN REASONABLE REACH</li> <li>• WIDE RANGE OF ENGINES AVAILABLE</li> </ul>	<ul style="list-style-type: none"> <li>• MINIMUM COOLING REQ'MTS</li> <li>• LOWEST SPECIFIC WEIGHT</li> </ul>	<ul style="list-style-type: none"> <li>• BEST BSFC</li> <li>• CAN ACCEPT HIGH TEMP INLET AIR</li> </ul>	<ul style="list-style-type: none"> <li>• GOOD BSFC &amp; SPECIFIC WT</li> <li>• COMPACT</li> <li>• MULTI FUEL</li> <li>• GOOD TURBO COMPATIBILITY</li> <li>• STRONG NASA R&amp;D FOCUS</li> <li>• FEW PARTS</li> </ul>
MAJOR DISADVANTAGES	<ul style="list-style-type: none"> <li>• REQUIRES CONSIDERABLE INTERCOOLING FOR TURBOS</li> <li>• ENGINE COOLING IS HIGH</li> </ul>	<ul style="list-style-type: none"> <li>• EXCESSIVE FUEL CONSUMPTION AT PART POWER</li> <li>• LEAN FLAMMABILITY PROBLEMS</li> </ul>	<ul style="list-style-type: none"> <li>• SCAVENGING AIR REDUCES ENERGY AVAIL. TO TURBO</li> <li>• LARGE TURBO SYSTEM REQ'D</li> <li>• HIGH INLET PRESS REQ'D</li> <li>• AVAILABILITY POOR</li> </ul>	<ul style="list-style-type: none"> <li>• RELATIVELY HIGH INLET PRESS REQ'D</li> </ul>

<sup>(1)</sup> NO GEAR BOX OR PROP

TABLE 5  
ADVANCED TECHNOLOGY ENGINES (~ 400HP AT 70,000 FT)

NEAR TERM INCLUDING DEMO (1-5 YEARS)

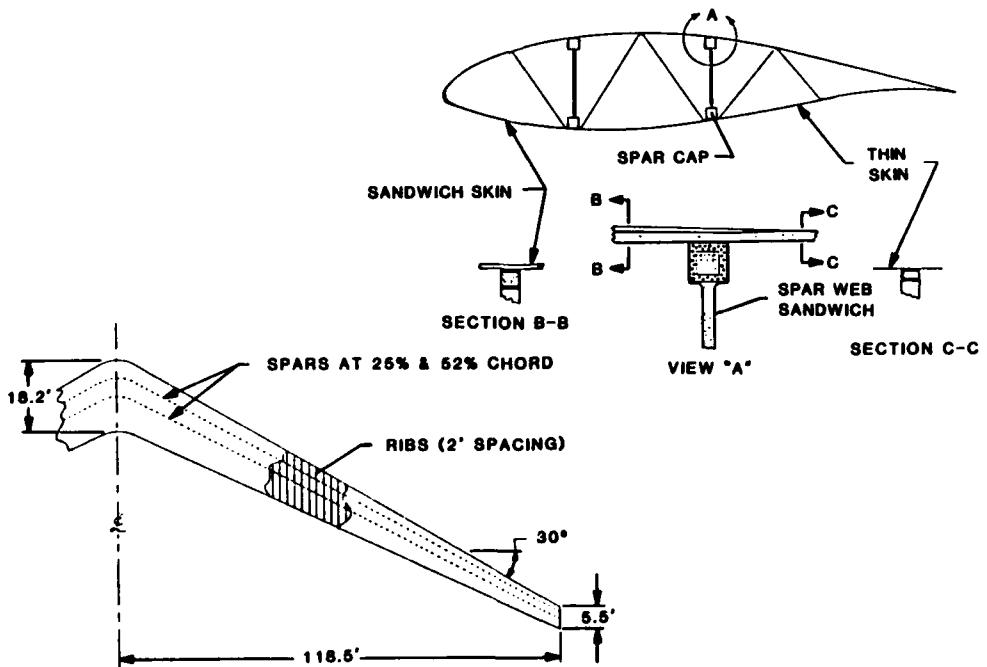
- SPARK IGNITION RECIPROCATING ENGINE
- TWO STAGE TURBOCHARGING SYSTEM
- TURBINE DRIVEN PUMPS AND FANS FOR COOLING
- BOTTOMING RANKINE CYCLE FOR ENGINE HEAT REJECTION AND AVIONIC POWER GENERATION
- WING RADIATORS FOR TURBO INTERCOOLING & RADAR

FAR TERM (5-10 YEARS)

- ROTARY ENGINE (TURBO COMPOUNDED)
- THREE STAGE TURBOCHARGING SYSTEM
- ELECTRICALLY DRIVEN PUMPS AND FANS FOR COOLING
- BOTTOMING RANKINE CYCLE FOR ENGINE HEAT REJECTION AND POWER GENERATION
- TURBO INTERCOOLING & RADAR COOLED WITH REFRIGERATION CYCLE BACK TO BACK WITH BOTTOMING RANKINE CYCLE

TABLE 6  
RECOMMENDATIONS

• GROSS WEIGHT	17,000 LBS
• MAX LIMIT LOAD FACTOR	3.0g
• SAFETY FACTOR	1.25
• RADAR TRANSPARENCY	
• STRUCTURAL WEIGHT FRACTION	17%
• FUEL WEIGHT	9540 LBS
• STRUCTURAL STABILITY	

TABLE 7  
DESIGN CRITERIAFIGURE 12  
STRUCTURAL CONFIGURATION

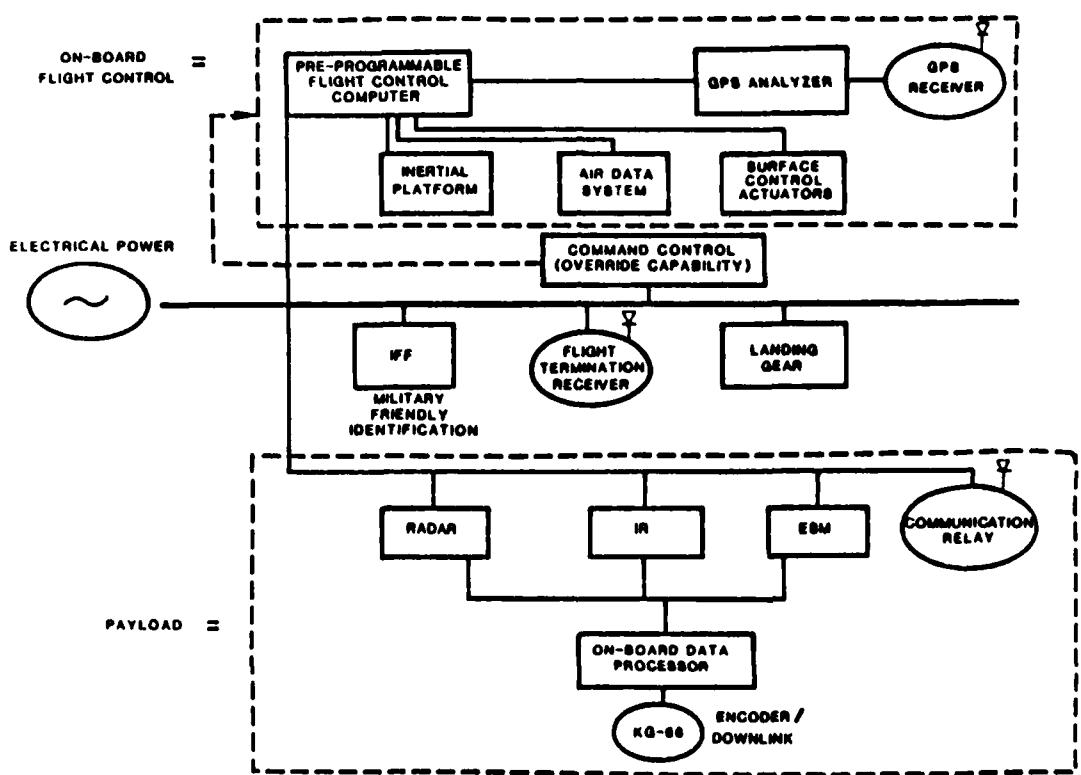


FIGURE 13  
AVIONICS/FLIGHT CONTROL/SENSORS

REPORT DOCUMENTATION PAGE				
1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document	
	AGARD-CP-388	ISBN 92-835-0390-2	UNCLASSIFIED	
5. Originator	Advisory Group for Aerospace Research and Development North Atlantic Treaty Organization 7 rue Ancelle, 92200 Neuilly sur Seine, France			
6. Title	GUIDANCE, CONTROL AND POSITIONING OF FUTURE PRECISION GUIDED STAND-OFF WEAPONS SYSTEMS			
7. Presented at	the Guidance and Control Panel 41st Symposium held at Ottawa, Canada, 8-11 October, 1985.			
8. Author(s)/Editor(s)	Various		9. Date June 1986	
10. Author's/Editor's Address	Various		11. Pages 160	
12. Distribution Statement	This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.			
13. Keywords/Descriptors	Air to surface missiles Missile guidance Command and control Weapon systems Navigational aids			
14. Abstract	<p>This volume contains 11 out of the unclassified papers presented at the Guidance and Control Panel 41st Symposium held in Ottawa, Canada from 8 to 11 October 1985.</p> <p>The classified papers are included in the publication CP 388 (Supplement). The papers were presented under the following headings: Tactical operational requirements; Weapon systems concepts; Design criteria and integration issues; Tactical mission planning and management, communication; Sensor aspects for guidance and control, position and navigation; Computational techniques and data processing; System demonstrations.</p>			

<p>AGARD Conference Proceedings No.388 Advisory Group for Aerospace Research and Development, NATO GUIDANCE, CONTROL AND POSITIONING OF FUTURE PRECISION GUIDED STAND-OFF WEAPONS SYSTEMS Published May 1986 160 pages</p>	<p>AGARD-CP-388 Air to surface missiles Missile guidance Command and control Weapon systems Navigational aids</p>	<p>This volume contains 11 out of the unclassified papers presented at the Guidance and Control Panel 41st Symposium held in Ottawa, Canada from 8 to 11 October 1985.</p> <p>The classified papers are included in the publication CP 388 (Supplement). The papers were presented under the P.T.O</p>	<p>AGARD Conference Proceedings No.388 Advisory Group for Aerospace Research and Development, NATO GUIDANCE, CONTROL AND POSITIONING OF FUTURE PRECISION GUIDED STAND-OFF WEAPONS SYSTEMS Published May 1986 160 pages</p>	<p>This volume contains 11 out of the unclassified papers presented at the Guidance and Control Panel 41st Symposium held in Ottawa, Canada from 8 to 11 October 1985.</p> <p>The classified papers are included in the publication CP 388 (Supplement). The papers were presented under the P.T.O</p>

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ISBN 92-835-0390-2

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